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BuAer Report AE-61-4

Fundamentals of Design
of Piloted Aircraft
Flight Control Systems

Volume V

THE ARTIFICIAL FEEL SYSTEM

PUBLISHED BY DIRECTION OF
THE CHIEF OF THE BUREAU OF AERONAUTICS

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MAY 1953

IMPORTANT NOTE

This volume was written by and for engineers and scientists who are concerned with the analysis and synthesis of piloted aircraft flight control systems. The Bureau of Aeronautics undertook the sponsorship of this project when it became apparent that many significant advances were being made in this extremely technical field and that the presentation and dissemination of information concerning such advances would be of benefit to the Services, to the airframe companies, and to the individuals concerned.

A contract for collecting, codifying, and presenting this scattered material was awarded to Northrop Aircraft, Inc., and the present basic volume represents the results of these efforts.

The need for such a volume as this is obvious to those working in the field. It is equally apparent that the rapid changes and refinements in the techniques used make it essential that new material be added as it becomes available. The best way of maintaining and improving the usefulness of this volume is therefore by frequent revisions to keep it as complete and as up-to-date as possible.

For these reasons, the Bureau of Aeronautics solicits suggestions for revisions and additions from those who make use of the volume. In some cases, these suggestions might be simply that the wording of a paragraph be changed for clarification; in other cases, whole sections outlining new techniques might be submitted.

Each suggestion will be acknowledged and will receive careful study. For those which are approved, revision pages will be prepared and distributed. Each of these will contain notations as necessary to give full credit to the person and organization responsible.

This cooperation on the part of the readers of this volume is vital. Suggestions forwarded to the Chief, Bureau of Aeronautics (Attention AE-612), Washington 25, D. C., will be most welcome.

L. M. Chattler
Head, Actuating & Flight Controls Systems Section
Airborne Equipment Division
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PREFACE

This volume has been written under BuAer Contract NOas 51-514(c) to present to those concerned with the problems of designing integrated aircraft controls systems certain information regarding the artificial feel system.

The purposes of this volume are to present the fundamental concepts underlying the design of artificial feel systems and to present a method of accomplishing this design so that the complete piloted aircraft system will meet certain specified requirements. Several basically new ideas have been included in this presentation.

The foremost of these is the inclusion of the human pilot in the analysis of the complete system. The validity of this procedure is based on the fact that the human pilot is an integral and essential part of any piloted aircraft system and that his opinion usually determines the acceptability of an airplane.

A second new concept presented is the re-definition of "the artificial feel system." In this volume, the artificial feel system includes not only the commonly accepted artificial force producing devices, such as bobweights, q-bellows, and centering springs, but also some subsystems, such as motion stability augmenters and autopilots. These subsystems must be included in the definition because they alter the static and dynamic stability, and hence the handling qualities and feel characteristics, of an airplane.

Special mention should be made of the following people for their help and cooperation: F. B. Bacus for coordinating the preparation and

publication of the volume, R. E. Gaskill for his work in transcribing the equations, and Shirley M. Keys and Dorothy L. Emerick for typing the manuscript.

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TABLE OF CONTENTS CONFIDENTIAL

CHAPTER I

INTRODUCTION

A modern high-speed, high-performance, piloted aircraft must be considered as a system when the problems of flight path control and stability are discussed. The concept of flight path control and stability can be easily visualized if the airframe motion is imagined as the motion of a velocity vector having both magnitude and direction.

Stability is determined by how well the airframe resists changes in the magnitude and direction of the velocity vector. On the other hand, control is a function of how well the velocity vector may be altered. In short, flight path stability is associated with the ease with which a pilot flies an airplane steadily, and flight path control is linked with the ease and precision with which a pilot maneuvers the airplane.

The controlled element in the pilot-airframe system is the basic airframe. Once the configuration of the basic airframe is determined, its characteristics are unalterable. The problem then is to design the controlling elements so that they act on the controlled element to give the desired system response.

These controlling elements are the human pilot and the airframe artificial feel system. The human pilot acts both as a sensing and an actuating element. Although the dynamic characteristics of pilots vary, the systems designer cannot vary these characteristics at his own discretion. The pilot therefore is an unalterable controlling element as far as the designer is concerned.

The other element in the pilot-airframe system is the artificial feel system which is defined as the combination of both the force producing system and the motion stability augmenting system.

The need for developing the artificial feel system can be traced directly to ever-increasing speeds and performance of aircraft. Few insurmountable problems of dynamic stability and control were encountered in the low-speed airplanes of a few decades ago. Any problems that did exist were solved by changing the basic airframe configurations.

As the speed range was extended, the designers began to resort to the use of spring tabs, horn balances, set-back control surfaces, and other aerodynamic devices. However, the effects of these modifications diminished at higher speeds as the dynamic pressures increased and the centers of pressure of the control surfaces moved aft. To enable the pilot to overcome the higher aerodynamic forces, it became necessary to use partially and fully-powered control surfaces.

With powered controls, part or all of the control feel to the pilot is lost unless artificial force producers are used. Since pilots normally fly by the physical association of applied force and maneuvering response, the need for force producers at the cockpit controls became critical and brought about the development of force stability augmenting systems.

The force stability augmenter has three basic purposes. First, it must provide the pilot with the proper pressure cues to allow near optimum flight path control. Second, it must aid in reducing the possibility of inadvertent destruction of the airplane. And third, the control surface motions produced by the force producer under hands-off flight conditions must result in satisfactory dynamic airplane stability.

The device used to aid in satisfying this third requirement is the motion stability augmenter, which may be defined as a system which automatically produces control surface movements that tend to increase the dynamic stability of the airplane. The airframe alone is not always attitude stable; that is, the

airframe motion may exhibit diverging or undamped oscillations or diverging exponentials. The pilot can prevent this type of motion by continually jockeying the cockpit controls. Obviously, this procedure will distract the pilot's attention from his other tasks and lower his efficiency.

The airplane may be stabilized by employing outboard or inboard stabilization. Outboard stabilization is a method by which the basic airframe configuration is altered, e.g., by changing wing dihedral or stabilizer surface areas. This method is objectionable because it may possibly lead to loss of control and increased drag effects or because it may be incompatible with the airplane performance requirements.

When inboard stabilization is used, various airframe motions are detected by appropriate sensors. The signals from these sensors are used to actuate motors or hydraulic servos which automatically deflect the control surfaces to counteract any undesirable airframe motions. By the prudent choice of sensors and actuators, the airframe can be made highly stable without loss of control.

The artificial feel system is relatively alterable, especially when fully-powered, irreversible controls are used. Thus the control system designer is faced with the task of achieving the desired complete pilot-airframe system response by a prudent design of the artificial feel system.

Chapter II of this volume is presented to familiarize the reader with the artificial feel problem. The factors that influence feel and the manner in which they affect the system response are discussed. In addition, present methods for supplying artificial feel to the pilot are briefly discussed.

Chapter III is devoted to the presentation of an analytical procedure for designing artificial feel systems to meet the piloted aircraft specifications summarized in Chapter IV.

An appendix presents the derivation of the augmented coefficients of the longitudinal characteristic equation of Chapter III.

CHAPTER II

THE CONTROL FEEL PROBLEM

SECTION 1 - INTRODUCTION

Because the concept of control feel has been gradually developed from opinions expressed by large numbers of pilots flying many types of aircraft, it is understandable that this concept is somewhat nebulous. This is especially true at present, for many of the old and established measuring sticks of desirable control feel and airplane stability appear to be losing significance for today's high-speed aircraft.

One of the first questions that arise in dealing with the control feel problem is a definition of the term "control feel." Consider the block diagram of the pilot-equivalent airframe system, shown in Figure II-1. The equivalent airframe includes the basic airframe plus any artificial feel system. Using this block diagram as a basis, control feel can be expressed as the ratio of the equivalent airframe response to the pilot's force input.

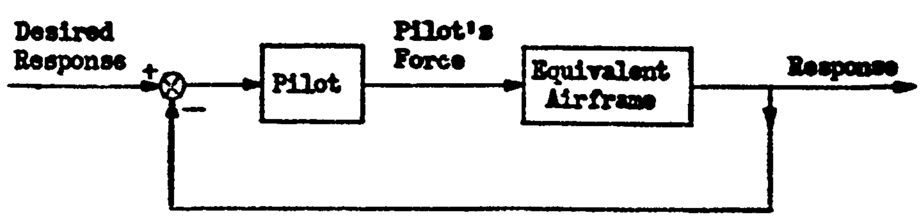


Figure II-1. Pilot-Equivalent Airframe System

Referring to Figure II-1, it is seen that control feel criteria expressed in this manner are apparently concerned with the equivalent airframe block only, and do not take into account the closed loop pilot-equivalent airframe system. Actually, however, desirable numerical values given in conjunction with these control feel criteria are normally obtained from a pilot-aircraft combination

and will therefore be applicable to the closed loop pilot-equivalent airframe system. For this reason, control feel criteria in this volume will be concerned only with applied forces and associated responses, but will implicitly involve the human pilot.

One shortcoming of this particular concept of control feel criteria is that no consideration is given to the deflection of the cockpit controls. In other words, it is assumed that the pilot flies by force feel only. Although it is recognized that the amount of deflection of a control is certainly a factor in the control feel problem, there has been insufficient correlation of data to evaluate the importance of this factor at the present time.

Various specific criteria for evaluating control feel have been proposed through the years. The most consistently named of these criteria, reflecting the majority of pilots' opinions, have been compiled into the flying qualities specifications published by the Navy Bureau of Aeronautics, the Air Forces, and the Civil Aeronautics Administration. Because these criteria have evolved from a large amount of flying experience, they have a fairly sound basis and should not be underrated. On the other hand, because the equivalent airframe block is always in a process of rapid evolution, the criteria for good control feel are subjected to frequent modifications. In fact, because of the recent accelerated development of the high-speed jet airplane with its associated systems — power control systems, artificial feel devices, and electronic stability augmenters — it appears that the entire concept of control feel and its various criteria will undergo radical changes.

Before attempting to evaluate the concept of control feel, it is important as background material to be familiar with the general criteria and fundamental aspects which have already been established from flying experience. Section 2,

therefore, gives a qualitative discussion of these fundamental aspects, which may be described as what the pilot wants for desirable feel characteristics.

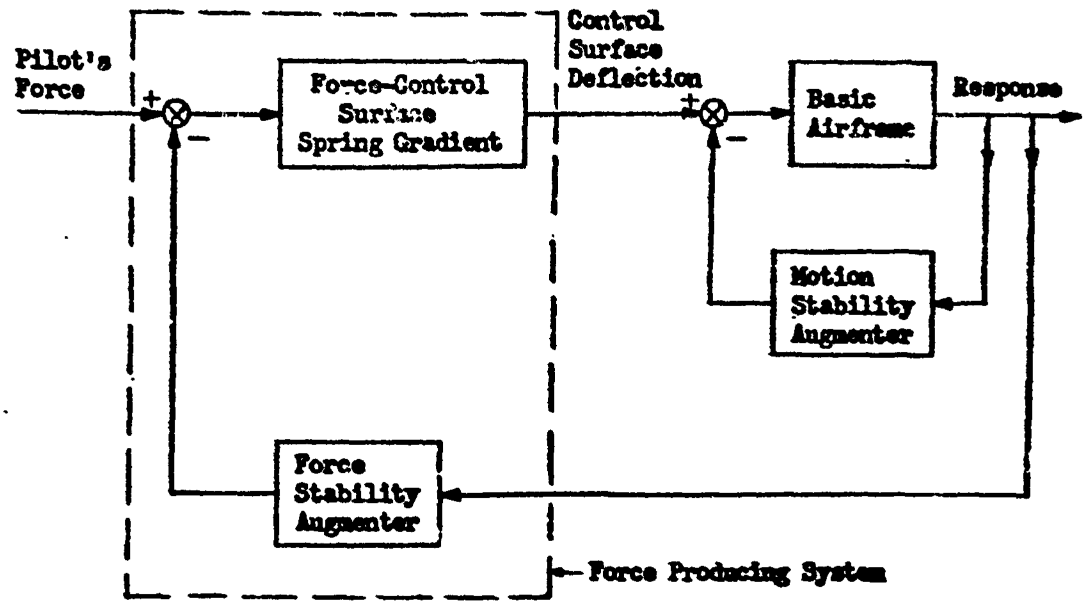


Figure II-2. Equivalent Airframe Block Breakdown

In Sections 3 and 4, the equivalent airframe block of Figure II-1 is broken down into its component blocks (see Figure II-2), and these are discussed separately. A general discussion of the basic airframe block is given in Section 3 along with the various factors that affect the basic airframe response to control surface deflections. It is shown how the wide variation of the response over the flight regime of an airplane creates the majority of the control feel problems. In Section 4, some of the more common elements in present-day force producing systems are discussed. The motion stability augmenter block is treated in detail in Sections 2 and 4 of Chapter III.

SECTION 2 - FUNDAMENTAL ASPECTS

As stated before, control feel qualities are measured by the response of the airplane to the pilot's control input. Nearly all pilots agree that the fundamental input parameter is force, and that they are not particularly concerned with the distance through which the force is moved. However, there have occurred recent instances where the control deflections have proved important. More will be said about this later when longitudinal control feel is discussed.

When the pilot applies a force input, he expects the airplane to respond in a certa . way. The desired response of the airplane is usually a steady state maneuver. For a constant applied elevator stick force the pilot expects a steady state normal acceleration response; for a constant applied aileron stick force he expects a steady state rolling velocity response; and for a constant applied rudder pedal force he expects a steady state sideslip angle response.

Furthermore, the pilot would like all these respective inputs and associated responses to act independently of each other; i.e., there should be no cross-coupling effects. For example, when he applies aileron control, he wants a rolling response with no sideslip. In turning maneuvers, the application of aileron and elevator controls only, which would be desirable from a pilot's viewpoint, will usually result in an uncoordinated turn if the inherent coupling between the rolling and sideslipping motions of the basic airframe is not eliminated. Although it is difficult to eliminate cross-coupling effects by aerodynamic means alone, these effects can be minimized by using automatic stability augmentation.

On the other hand, the pilot likes a certain degree of correlation among longitudinal, lateral, and directional control feels. For example, if the

elevator control forces are high, then the pilot wants the aileron control forces high. Very little work has been done toward the establishment of criteria to describe this "balance" of control feel.

(a) LONGITUDINAL CONTROL FEEL

Longitudinal control feel is usually treated as if it consisted of two types: the feel necessary for straight and level equilibrium flight and the feel necessary for maneuvering flight. Although these are often considered separately in design work, mainly because of the individual design criteria which have been established for each, it should be emphasized that it is the integrated effect which produces the longitudinal control characteristics felt by the pilot. In other words, the border line between these two types of longitudinal control feel is not well defined. Nevertheless, for analysis and discussion purposes, it is still convenient to consider the equilibrium flight and the maneuvering flight types separately.

In addition, longitudinal maneuvering flight may be divided into two types: steady state type maneuvers in which the value of normal acceleration is different from 1 g but once established remains essentially constant with time during the maneuver, such as in ordinary turns and dive recovery pull-outs; and transient type maneuvers in which the normal acceleration never reaches a steady state value, such as in abrupt pull-ups from level flight, and responses to pulse-type elevator inputs.

Although the various time-proved criteria which have been established for the steady state type maneuvers can be discussed with a certain degree of completeness and assurance, adequate criteria for the transient type maneuvers have not been established. Because of the increasing importance of the transient type maneuver for high-speed aircraft, more attention should be devoted to it.

For the purposes of this volume, then, there are three fundamental aspects and associated criteria of longitudinal control feel: (1) control feel which involves a change in speed from an original equilibrium speed and which is given primarily by the gradient of stick force per change in forward speed, dF_s/dV; (2) control feel which involves steady state normal acceleration and which is given by the ratio of stick force per change in normal acceleration, $\Delta F_s/\Delta n$; and (3) control feel which involves transient normal accelerations and for which no definite criteria have evolved as yet.

CHANGE FROM EQUILIBRIUM SPEED

When the forward speed of an airplane changes from an equilibrium speed, the pilot expects an accompanying change in elevator stick force. A typical plot of elevator stick force versus equilibrium speed is shown in Figure II-3. Each point along this curve represents the stick force necessary to change the airplane's equilibrium flight condition, assuming the stick force was trimmed ($F_s = 0$) at the original equilibrium speed. Notice that to increase the speed of the airplane, the pilot must exert a push force on the stick, and to decrease the speed, he must exert a pull force. The slope of this curve, dF_s/dV or dF_s/dM, is very important as a control feel parameter because it gives the pilot an indication of the static stability of the airframe. If the airplane is disturbed so that the speed is changed from the stick force trim speed, but the stick is left free, the push or pull forces to maintain the new speed are of course not applied, and if the stick force gradient is in a stable direction, the tendency to regain the trim speed results automatically. For this reason, the gradient of stick force versus speed is a measure of the so-called stick-free static stability of the airframe. A negative slope, such as that shown in Figure II-3, indicates a condition of stick-free static stability, and conversely, a positive slope indicates stick-free static instability.

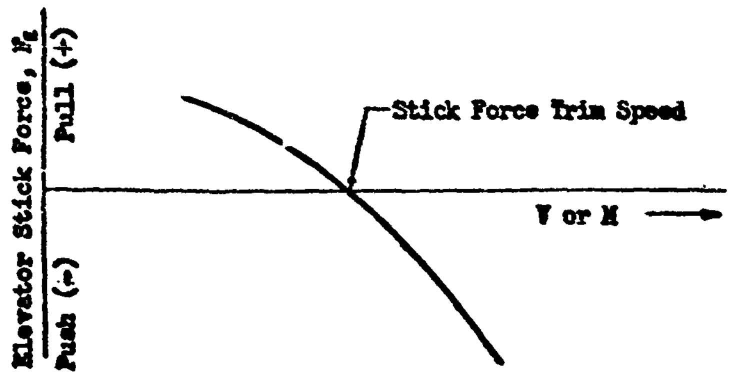


Figure II-3. Elevator Stick Force versus Equilibrium Speed

In general, a large negative value of dF_s/dV is desirable. A large gradient will tend to keep the airplane flying at constant speed, and is therefore especially helpful during gusty weather conditions. It will enable the pilot to trim the airplane easily and to maintain his desired trim speed with a minimum of effort. It also provides stall warning for low-speed flight. On the other hand, too large a gradient is undesirable for combat maneuvering because the variations in speed from an original force trim speed can become quite large, producing high stick forces which induce pilot fatigue. Also, when trimmed in a landing approach, too large a gradient may produce excessively high stick forces during the landing flare-out.

An important consideration in the longitudinal feel problem is the magnitude of the friction forces in the control system. These forces must be as small as possible so that the feel is not completely masked. Figure II-4 indicates the typical hysteresis loop caused by control system friction when the equilibrium speed of an airplane is first increased and then reduced from the original equilibrium speed by the pilot.

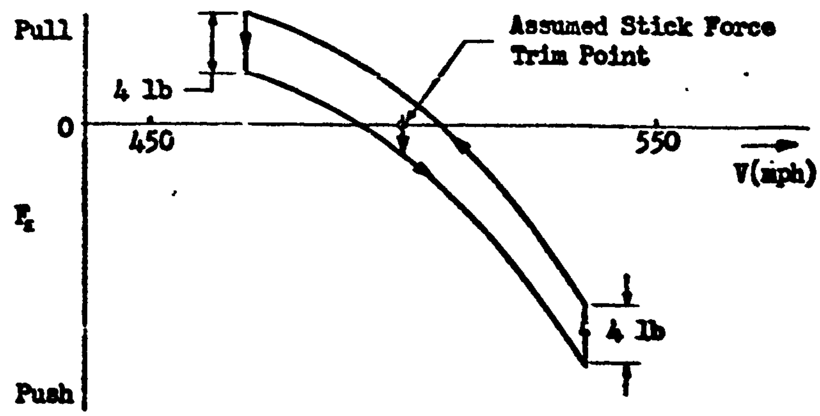


Figure II-4. Effect of Control System Friction on Elevator Stick Force versus Equilibrium Speed

For this case, the assumed friction force band of 4 pounds causes a "dead-band" of V of the order of 30 mph. This means that if the pilot has trimmed the stick forces for a particular equilibrium speed, it is possible for the speed to change 30 mph before any elevator stick force signifying this change is felt by the pilot.

These considerations concerning a desirable gradient, dF_s/dV, show that what pilots really need is a nonlinear gradient: a high gradient around the trim speed to alleviate the friction problem and to provide good feel, and a small gradient at speeds on either side of the trim speed to prevent excessively high stick forces for maneuvering and landing, and possibly another high gradient at speeds much lower than trim speed to reduce the possibility of a stall, as shown in Figure II-5.

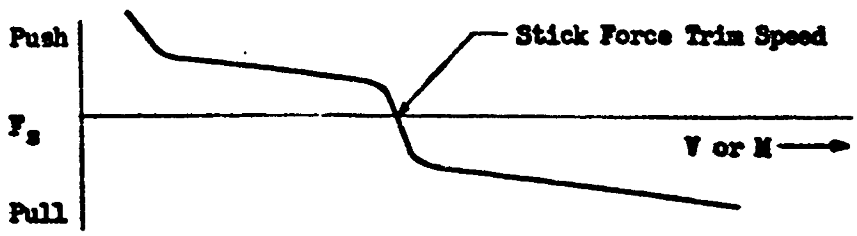


Figure II-5. Desirable Nonlinear Characteristics of Elevator Stick Force versus Equilibrium Speed

Another important aspect of the longitudinal feel problem which is often taken for granted is that a change in elevator stick position always accompanies a change in elevator stick force as the forward speed is changed from an original equilibrium speed. In other words, it is assumed that a forward stick deflection is associated with a push force, and an aft stick deflection is associated with a pull force. The effects of stick deflection on the pilot's opinion of longitudinal feel are not known exactly at present and should be investigated.

STEADY STATE MANEUVERS

In addition to the longitudinal feel required for straight and level equilibrium flight, it has been found by experience that longitudinal feel is very important in steady state maneuvering flight where stabilized values of normal accelerations different from 1 g are involved.

The two most common instances of this type of maneuvering flight are ordinary turns and dive recovery pull-outs. For these maneuvers, a more or less constant value of normal acceleration is maintained, and for this reason, feel criteria in longitudinal maneuvering flight in the past have been based on these steady state values of normal acceleration as the independent variables.

Figure II-6 gives a typical plot of elevator stick force versus steady state normal acceleration. The slope of this curve is the gradient of stick force per g. This gradient has been found to be closely related to the pilot's opinion of the maneuvering stability of an airplane.

In Figure II-6, notice that a pull force is required on the stick to produce a positive increase in normal acceleration. This is a stable stick force per g gradient, and the airplane is said to possess stick-free maneuvering stability because if the pilot does not apply the necessary force to maintain the normal acceleration increment, the stick tends to move in a direction which

causes the airplane to return to a 1 g flight condition. It can be seen that a stable gradient is highly desirable for flight in gusty weather.

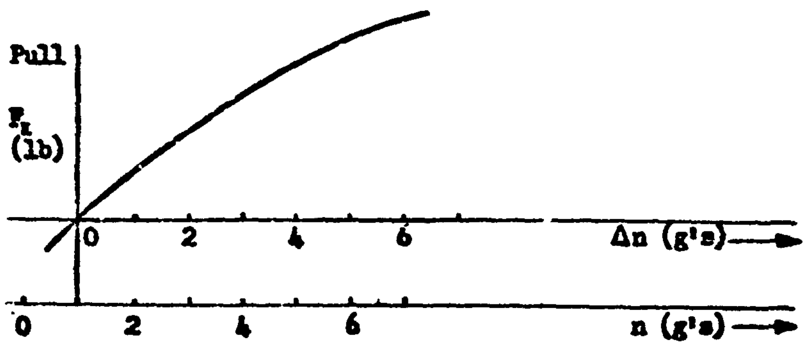


Figure II-6. Elevator Stick Force versus Normal Acceleration in Steady State Maneuvers

It is very important not only to provide a stable stick force per g gradient but also to keep this gradient within certain desirable limits. Stick force per g gradients that are too high will make the airplane feel sluggish and will reduce the combat effectiveness of the pilot-airplane combination. Gradients that are too low will make the airplane seem too sensitive, and will induce excessive pilot fatigue in gusty weather flights since the pilot will have to "fly" the airplane continuously. In addition, low gradients provide insufficient warning to the pilot that structural limits are being approached during maneuvering flight.

For steady state type longitudinal maneuvers, it is apparent that what the pilots desire is a nonlinear stick force per g gradient; that is, a steep gradient near 1 g to provide the necessary sensitivity and gusty weather stability, a low gradient for the intermediate g range to prevent pilot fatigue in combat maneuvering, and finally a steep gradient in the high g range to provide adequate structural limitation warning as shown in Figure II-7.

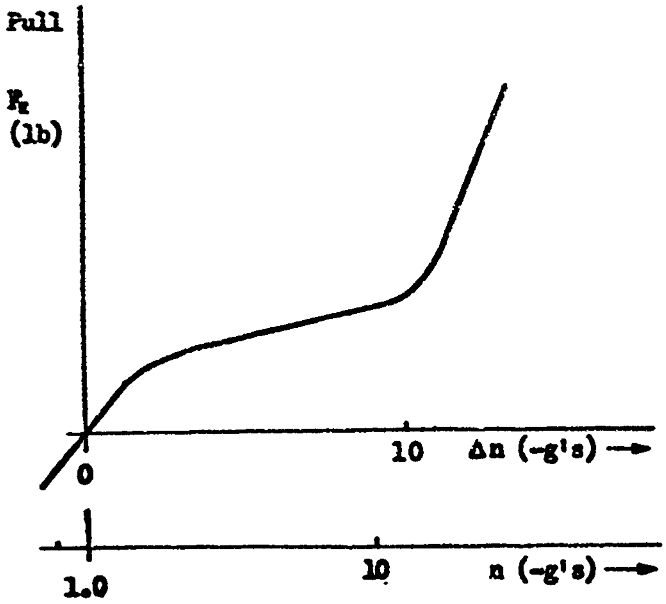


Figure II-7. Desired Elevator Stick Force versus Normal Acceleration in Steady State Maneuvers

TRANSIENT MANEUVERS

Longitudinal control feel in transient maneuvers was never considered very important in the past except for a few isolated cases of airplanes which developed undesirable transient characteristics. Today, however, mainly due to high speed flight, various aerodynamic artifices and mechanical devices are necessary to produce satisfactory stability and control. It has been found in many cases that although the steady state maneuvering control feel characteristics of an airplane may satisfactorily meet the required design values, the airplane may exhibit unsatisfactory feel characteristics in transient maneuvers.

In many of these cases the causes of unsatisfactory transient feel are not apparent, and flight test records have not helped in determining these causes.

However, it can be stated with some assurance that pilots desire a higher relative value of stick force in rapid pull-up maneuvers that in slow pull-up maneuvers.

Figure II-8 compares typical time histories of an airframe with good transient feel characteristics and one with poor transient feel characteristics. Notice that in the latter case, the maximum value of stick force is less in the abrupt pull-up than in the slower pull-up.

This can become important when considering structural limitations of an airplane. In transient maneuvers the pilot requires some sort of feel indication which will tell him the magnitude of the load factor which will ultimately be attained in the transient. If there is insufficient warning in the feel characteristics, the structural limit load factor of the airplane can easily be exceeded by the pilot.

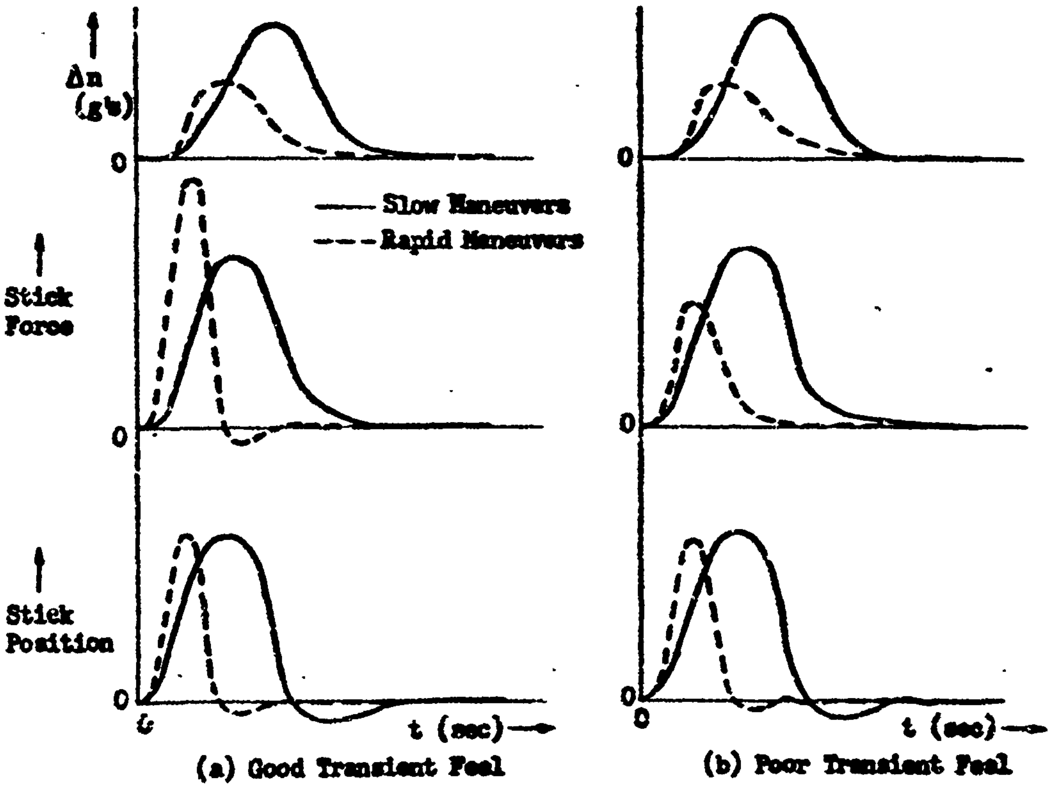


Figure II-8. Control Feel in Transient Longitudinal Maneuvers

Most of the difficulties illustrated in Figure II-8 apparently arise because too large a portion of the maneuvering stick force comes from a source — such as a bobweight — which is in phase with the normal acceleration of the airplane. Since there can be an appreciable time lag in the normal acceleration response to an abrupt stick input, the portion of the force feel in phase with the normal acceleration can therefore lag the stick motion, thus creating poor feel characteristics.

Another aspect of the transient feel problem which is often overlooked is the amount of stick deflection required in rapid maneuvering. Experimental data indicate that pilots expect a certain amount of stick deflection when executing transient maneuvers, and that deflections below a certain minimum value are considered undesirable. However, just what criterion is to be used in establishing this minimum stick deflection value is not known.

This leads to the problem of setting up suitable criteria for transient feel. Although there is no reason to question the validity of the criterion implied in the preceding discussion, i.e., the maximum stick force should be a function of the rapidity of the maneuver, this criterion does not take into account the amount of stick deflection. Hence, it would appear either that this criterion should be extended to include stick deflection, or that additional criteria may be necessary. Various proposals have been suggested which take into account the work or the power applied to the stick as functions of the transient load factor response attained. These proposals recognize stick displacement and rate of displacement as factors in transient feel, but these criteria are as yet untested.

(b) LATERAL CONTROL FEEL

Lateral control feel is concerned with rolling motion such as that used in performing ordinary banked turns and certain combat maneuvers. In these lateral maneuvers, the pilot expects a certain stick force and an associated rolling

response when he displaces the control stick sideways. Past experience has indicated that this desired response is rolling velocity rather than roll angle; more will be said about this later.

If a constant applied aileron force is maintained, most airplanes will attain a more or less steady state value of rolling velocity, as shown in Figure II-9. Notice that the curve is nonlinear and that disproportionately higher stick forces are necessary to produce the larger rolling velocities. This is undesirable because the maximum available sideways force input from the pilot is quite limited. In fact, this is the main reason for using a wheel type control column in large aircraft. However, wheel controls are not very satisfactory for fighter type aircraft because of space limitations. It then appears that a nonlinear variation of aileron force versus steady state rolling velocity is desirable, but that the variation should be similar to that shown in Figure II-10. Here, the force gradient is high around neutral to provide adequate feel characteristics including sufficient centering tendencies to overcome control system friction. The gradient is kept low in the region of high rolling rates so that the forces required to maneuver do not exceed the pilot's capabilities.

It should be pointed out that the criterion for lateral response is not usually expressed in terms of pure rolling velocity, but in terms of a non-dimensional rolling velocity, $pb/2V$, which can be thought of as the wing tip helix angle, i.e., the flight path angle through which the tip of the wing moves during a steady state rolling motion. This non-dimensional rolling velocity is used because the rolling response characteristics of airplanes are inherently a more direct function of this helix angle than of pure rolling velocity.

In the past, the lateral feel criterion has been based on a maximum value of aileron force required to attain the maximum available wing tip helix angle.

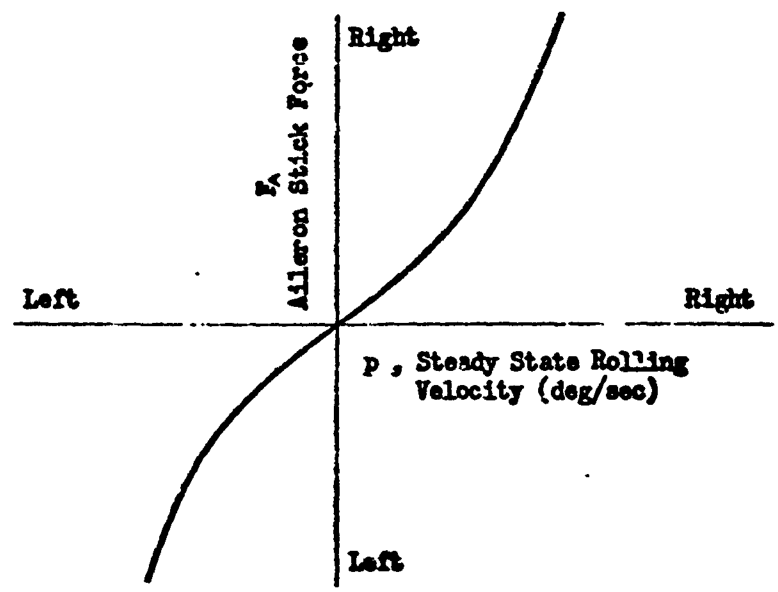


Figure II-9. Typical Aileron Stick Force versus Steady State Rolling Velocity

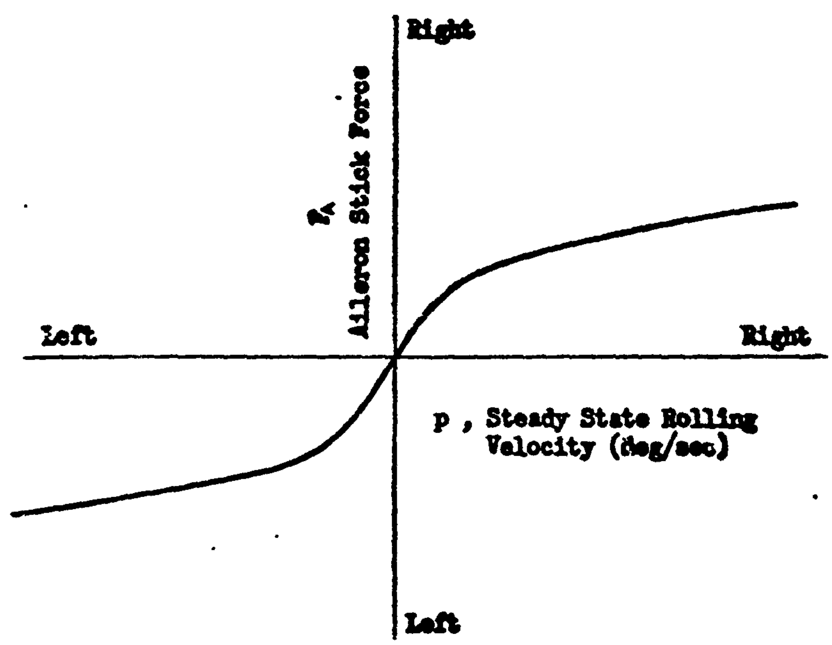


Figure II-10. Desired Aileron Stick Force versus Steady State Rolling Velocity

For the purpose of this volume, this criterion can essentially be given in terms of $F_A/(pb/2V)$, which implies a certain proportionality between applied aileron force and associated steady state rolling response.

Recently, it has been shown that the rolling response parameter, $pb/2V$, which is based on steady state rolling velocity, is not entirely satisfactory for modern high-speed aircraft. One reason is that smaller aircraft of low aspect-ratio wing planform never reach a steady state rolling velocity in any sort of maneuver. Another reason is that the non-dimensional wing tip helix parameter, $pb/2V$, apparently loses much of its significance as a design parameter when a modern high-speed airplane is considered because this parameter is based on the assumption that the rolling velocity, p, is directly proportional to the true airspeed, V. This assumption is an oversimplification which is essentially correct only for relatively low subsonic Mach numbers and for rigid airplanes. This is explained in greater detail in Section 3(b).

For the reasons given above, it is likely that new lateral response criteria will not be based on rolling velocity, but on time required to attain a certain roll angle. With this new rolling response criterion it is not clear what the basis for aileron stick force feel criteria should be. Until the new criteria can be established, it is recommended that $F_A/(pb/2V)$ be used, but modified to the extent that the helix angle response, $pb/2V$, should not be considered a steady state value but a maximum value which can be obtained with maximum available aileron deflection.

(c) DIRECTIONAL CONTROL FEEL

Directional control feel, in the popular use of the expression, is the sideslip response of the airplane to rudder pedal force input. The word "directional" in this sense should not be interpreted to mean the direction of the flight path

of the airplane, but rather to mean the direction in which the nose of the airplane is pointed with reference to its flight path. To change the flight path direction, the airplane is usually banked into a turn, which implies that the aileron is the primary directional flight path control.

One of the main purposes of the rudder control is to prevent the build-up of sideslip in turning maneuvers; in other words, the rudder is primarily used to keep the nose of the airplane directed along the flight path.

Because the rudder is a secondary control when compared to the elevator and aileron, the directional control feel problem is not as serious as the problem encountered in longitudinal or lateral control feel.

The physical setup of the directional control system further reduces the problems of directional control feel. It is conventional practice to use rudder pedals for directional control. Since the pilot's legs are very powerful, high directional forces are tolerable. Furthermore, since the legs are not very sensitive to small forces, centering springs can be successfully used in the rudder control system to mask the system friction.

For directional control feel, the pilot wants a steady state sideslip in response to a constant applied rudder pedal force. For positive directional stability, a right rudder pedal push force produces a left sideslip and conversely.

A typical curve of rudder pedal force versus steady state sideslip angle is shown in Figure II-11. When the slope of the curve is as shown in this figure, the airplane is said to possess rudder-free static stability because, if the pilot did not apply the rudder pedal force necessary to maintain the sideslip angle, the airplane would tend to return to zero sideslip of its own accord. Evidently, not very much importance has been placed on the value of the gradient of pedal force versus sideslip angle in the past. Pilots seem to be satisfied if the gradient is in the proper direction and if the maximum pedal

force is kept within their physical capability. However, it must be realized that gradients which are too high will induce pilot fatigue in combat maneuvering, and gradients which are too low create oversensitivity in directional control, making it difficult for the pilot to coordinate turns and perform precision maneuvers. Another important aspect is that a gradient which is too low will permit the pilot to inadvertently sideslip the airplane to large angles at high speeds, which may cause structural failure of the vertical tail.

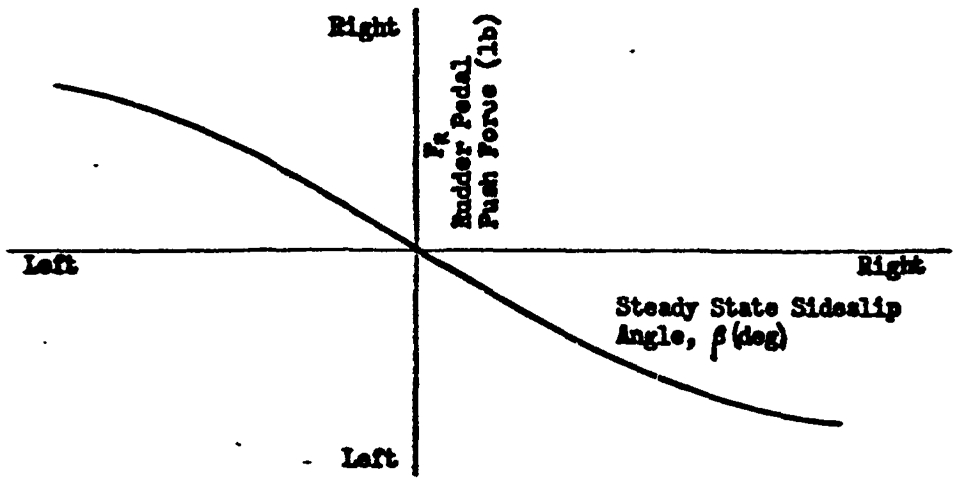


Figure II-11. Typical Rudder Pedal Force versus Steady State Sideslip Angle

Figure II-12 indicates a proposed desirable nonlinear curve of rudder pedal force versus steady state sideslip angle. A high gradient around zero sideslip is desirable for good centering characteristics. A less steep gradient is desirable for intermediate sideslip angles to delay pilot fatigue. A very high gradient at high sideslip angles is desirable to prevent inadvertent overstressing of the vertical tail.

In Figure II-12, the problem arises of selecting the proper value of β_1, the sideslip angle at which the high gradient begins. This high gradient

breakpoint should probably occur at low values of β when the airplane is flying at high dynamic pressures because the vertical tail load is proportional to the product of dynamic pressure and β. Consequently, a possibly better abscissa to be used in Figure II-12 is $q\beta$, rather than β alone.

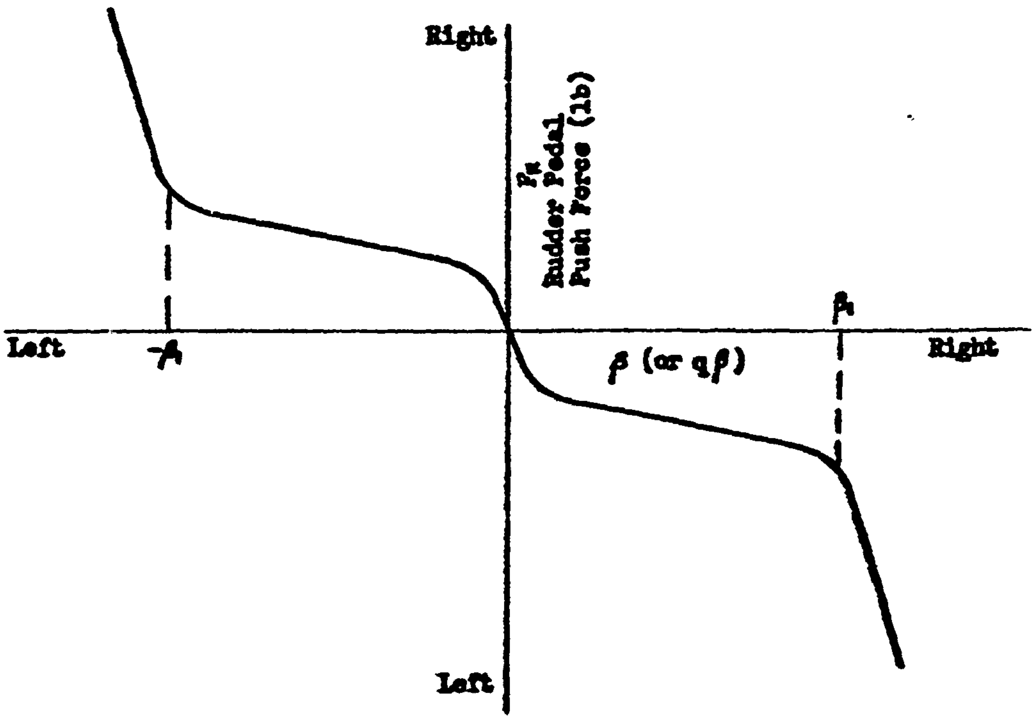


Figure II-12. Desirable Rudder Pedal Force versus Steady State Sideslip Angle

SECTION 3 - FACTORS AFFECTING AIRFRAME RESPONSE

In this section the basic airframe block will be discussed. This is the most important of all the blocks making up the control feel system. Most of the problems of control feel design arise because of the various inherent characteristics of the basic airframe. It is not profitable to go into great detail because each airplane differs from the others and has its own particular characteristics.

The purposes of this section are to point out some of the important factors which affect the airframe response and to show how these factors can produce wide variations in response over the flight regime of any given airplane.

The discussion to follow concerns the airplane response to control deflections, and does not concern control forces. Feel forces are not discussed at this point because these forces depend not only on the airplane response but also on the particular elements comprising the feel system, thus creating far too many variables for a general discussion. However, the control force response curves can be visualized if it is assumed that a simple spring gradient relation exists between control deflections and control forces.

(a) LONGITUDINAL RESPONSE

ELEVATOR ANGLE VERSUS EQUILIBRIUM SPEED

One of the most troublesome aspects of the stick force versus equilibrium speed problem is that an unstable gradient of elevator angle to trim occurs over a certain Mach number region for most present high-speed aircraft. Figure II-13 shows a curve of elevator angle to trim versus Mach number for a typical swept wing airplane configuration.

The unstable slope occurs in the transonic region and is associated primarily with the aft shift of center of pressure on the wings.

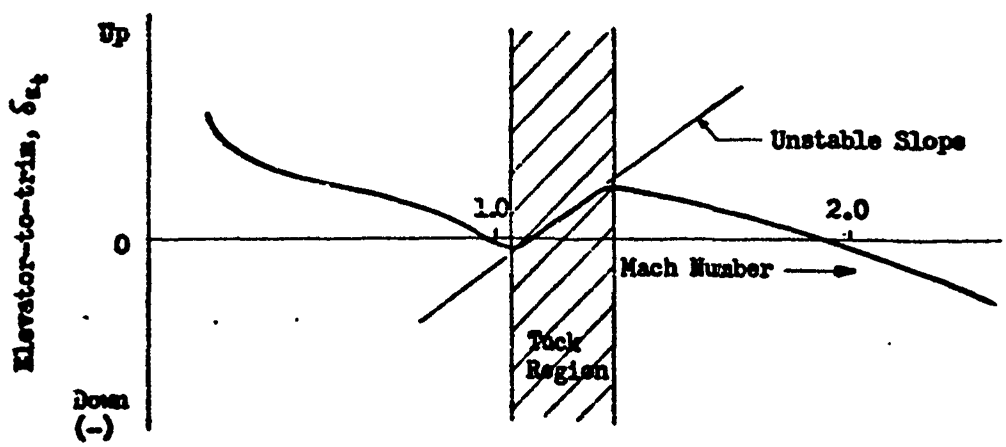


Figure II-13. Typical Curve of Elevator to Trim versus Mach Number

The degree of instability depends upon the wing and tail configuration of the particular airplane, but in general, it is more severe for straight wing airplanes and is less severe for delta wing aircraft. In most cases, it appears that the lower the aspect ratio of the wing, the less severe the degree of the unstable slope. Pilots usually refer to this unstable slope region as the "tuck" region. Consider the case of increasing the trim Mach number in Figure II-13. Starting at a low Mach number, the pilot must apply more and more down-elevator (more elevator stick push force) to maintain level flight as the Mach number gradually increases. This leads to good control feel. However, near the tuck region, as the pilot continues to apply more down-elevator, the airplane will nose down and dive because the pilot is applying more down-elevator than is required to trim. This nosing down tendency is referred to as "tuck-under."

Now consider the case of decreasing the trim Mach number starting from some supersonic value. In this case the pilot must keep applying more up-elevator (more pull stick force) as the trim Mach number decreases until the tuck region is encountered. If the pilot maintains the up-elevator trend, the airplane will nose up and pull positive normal acceleration. This effect is referred to as "tuck-up." Therefore, it is seen that a "tuck-under" is associated with an increasing trim speed, and that a "tuck-up" is associated with a decreasing trim speed. The tuck region or region of unstable slope is of course undesirable, especially if the airplane is designed to cruise in this Mach region because the airplane will always tend to diverge from its trim speed.

However, pilots evidently tolerate this situation as long as the airplane possesses maneuvering stability, i.e., a stable stick force per g gradient. There is a possibility, however, that a combination of a severe tuck region and a low stick force per g gradient can be of serious consequence, for example, in the case of a dive recovery, where the Mach number is decreasing and the tuck-up region is suddenly encountered. Here the possibility of structural failure is

imminent because the airplane may be subjected to large positive normal accelerations.

Another aspect of the stick force versus equilibrium speed problem is the rapid rise in the slope of the elevator to trim curve for very low Mach numbers, i.e., for landing as shown in Figure II-13. This situation usually creates a control feel design problem in artificial feel systems if a mechanical spring is used to create stick force proportional to elevator deflection. Since large elevator deflections are necessary for landing, undesirably high landing stick forces will result.

ELEVATOR ANGLE PER CHANGE IN LOAD FACTOR

One of the main factors which affect the steady state maneuvering response, $\Delta\delta_E/\Delta n$, of a high-speed airplane is Mach number. Figure II-14 illustrates that a drastic increase in elevator angle per g occurs when passing from transonic to supersonic Mach numbers. This large increase is due to the combination of the increase in the inherent static stability of the airplane and the reduction of elevator control effectiveness. If the ordinate is in terms of stick force per g and if reasonable values of stick force per g are selected for the subsonic region, intolerably high values of stick force might occur in the supersonic region.

As shown in Figure II-14, the center of gravity position is important, but its relative effect is small in comparison with the Mach effect.

Although the basic trend of this curve with Mach number is essentially the same for all airplane configurations, the severity of the change in $\Delta\delta_E/\Delta n$ is definitely a function of the wing and tail configurations. Straight wing aircraft show the largest and most abrupt changes with Mach number; swept wing airplanes show less severe effects; and delta wing planforms show the least overall change, and in comparison with other wing configurations, this change occurs gradually with Mach number.

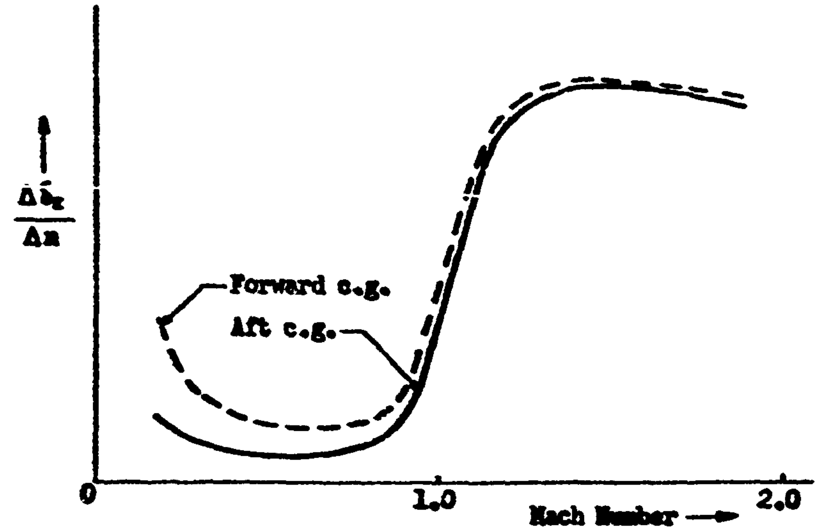


Figure II-14. Typical Curve of Elevator Angle per g versus Mach Number for Steady State Maneuvers

The tail configuration is one of the most important factors influencing $\Delta\delta_E/\Delta n$. The severity of the change of $\Delta\delta_E/\Delta n$ is greatly reduced for an all-movable tail configuration as shown in Figure II-15.

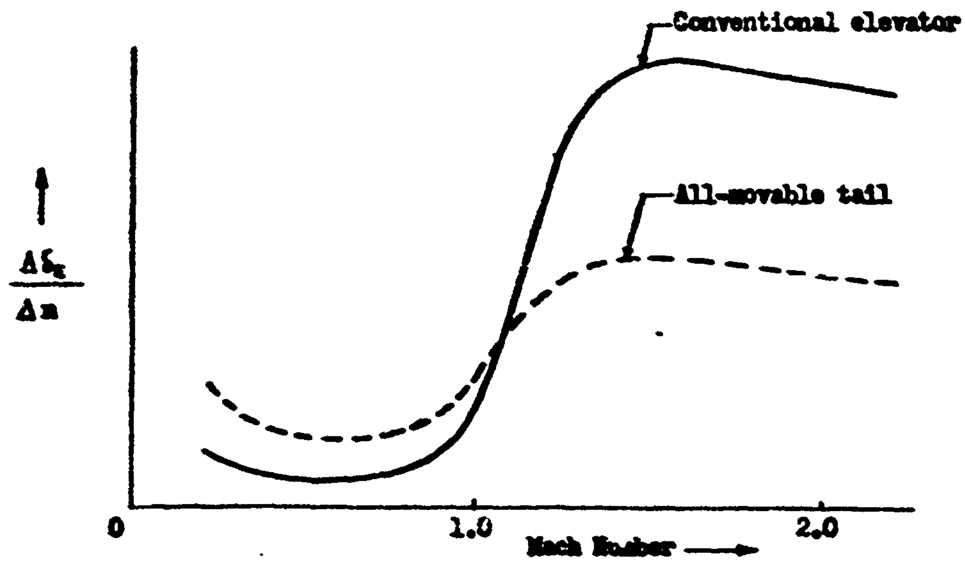


Figure II-15. Typical Comparison of Elevator (or Stabilizer) Angle per g versus Mach Number for Steady State Maneuvers

The large changes in elevator angle per g over the entire flight regime of a given airplane constitute one of the major problems of control feel design. This is especially significant because the longitudinal criterion of stick force per g is probably the most important criterion to be met for satisfactory feel characteristics.

Another aspect of the elevator angle per g problem is the rapid change in the slope of the curve for very low Mach numbers as shown in Figure II-14; i.e., large elevator deflections are required to produce load factor changes in this region. This creates a problem for any artifical feel system which has a mechanical spring as the force producer. However, the rapid rise in the $\Delta \delta_E / \Delta n$ curve at low Mach numbers is not as severe a problem as is the rapid rise in the elevator to trim curve since in practice it is seldom necessary or possible to pull much load factor at low speeds.

(b) LATERAL RESPONSE

The most important factors influencing lateral response characteristics are Mach number and aeroelasticity. Figure II-16 shows a curve of rolling velocity response versus Mach number for two altitudes. The same response is presented in two different forms: actual rolling velocity per unit aileron deflection in Figure II-16 (a) and non-dimensional rolling velocity (wing tip helix angle) per unit aileron in Figure II-16 (b). For each of these curves it is assumed that the aileron deflections are small enough that a steady state rolling velocity response is physically realizable.

These figures bring out the fact that there is an approximately linear increase in rolling velocity with Mach number in the low subsonic range. This effect originally prompted the definition of the non-dimensional rolling velocity parameter, $pb/2V$, which gives a single rolling response criterion for different airplane configurations. Notice that this linearity starts breaking

down in the transonic region, indicating that the $pb/2V$ parameter loses significance for a supersonic airplane, but only as far as this linearity is concerned. When comparing rolling responses of airplanes of different sizes at the same Mach number, $pb/2V$ is still significant.

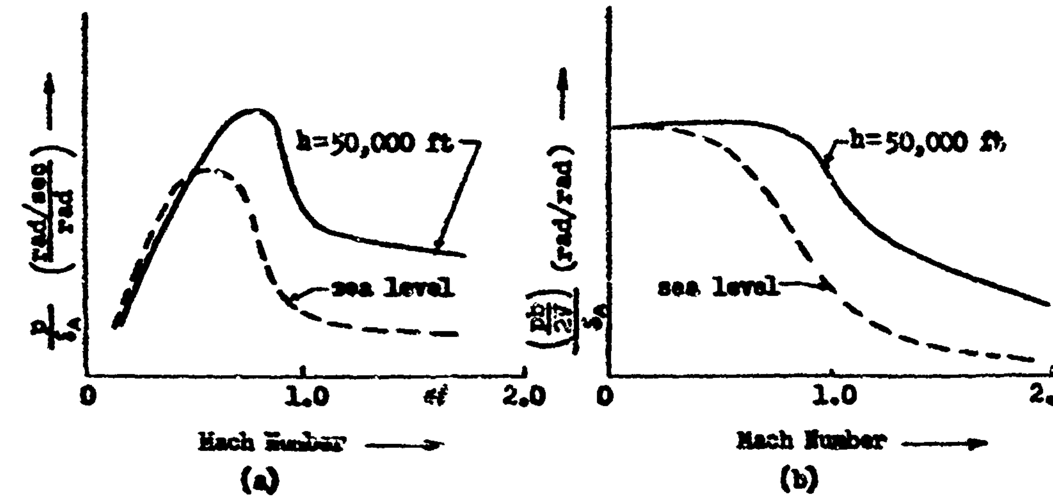


Figure II-16. Typical Rolling Response Curves for Two Altitudes

Notice the large decrease in rolling response in the transonic region and the further gradual decrease in the supersonic region. Notice also the large influence of aeroelasticity as shown by the different altitude curves.

The general trend of rolling response with Mach number, shown in Figure II-16, is essentially the same for any airplane configuration; however, the amount of decrease in the supersonic region and the severity of the transonic drop-off are apt to be less for low aspect ratio and delta wing configurations.

Another important factor which greatly affects lateral response characteristics is the power available to deflect the ailerons. High rates of deflection and relatively large deflections are necessary to meet satisfactorily the new rolling response criterion which gives the time to reach a given roll angle.

The response curves given in Figure II-16 implicitly assume that a steady state rolling velocity is obtainable and that this steady state velocity response is linearly proportional to the amount of applied aileron. However, under the new criterion the maximum attainable rolling velocity is the important parameter. Since maximum attainable rolling velocity is a direct function of maximum attainable aileron deflection, it is seen that control power available is a very important factor in lateral response characteristics. Figure II-17 shows the effect of available control power.

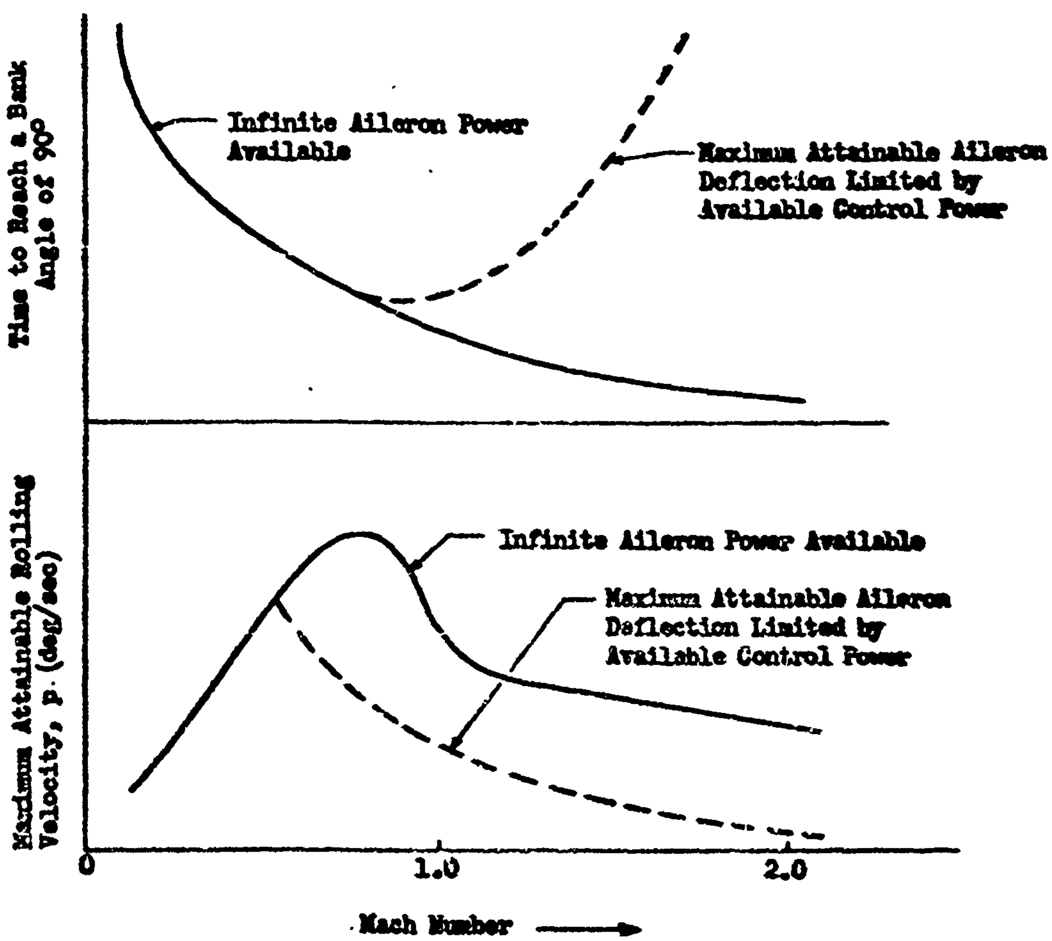


Figure II-17. Effect of Available Control Power to Aileron on Rolling Response of a Typical High-Speed Airplane

SECTION 4 - ARTIFICIAL FEEL DEVICES

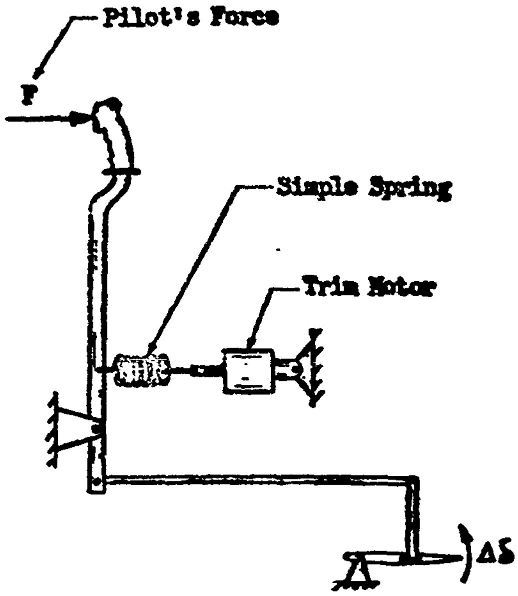
This section presents a brief discussion of some of the common artificial feel devices in use today. The discussion includes a short physical description of each device, a statement of its purpose, an account of how it affects the feel characteristics of a typical supersonic airplane in light of the feel criteria presented in Section II-2, and finally an appraisal of its limitations.

Most of these devices are used in "natural feel" systems as well as in fully-powered control systems. Natural feel systems are those in which all or part of the aerodynamic loads on the control surfaces are transmitted directly to the control stick and consequently are felt directly by the pilot. Fully powered control systems are those in which all control feel to the pilot is lost unless artificial feel devices are used.

The following illustrations of these various feel devices show them mounted on or near the control stick. This is done for illustrative purposes only; in actual practice, it may be better to mount these devices close to the control surface in order to minimize the problems of flexibility and backlash in the control feel system.

(a) SIMPLE SPRING

The most elementary force producer which can be used in artificial feel systems is the simple mechanical spring. Its purpose is to create a stick force proportional to control surface deflection. Figure II-18 (a) shows a schematic of a typical simple spring installation. The simple spring defined here does not include a preload. Extension or retraction of the trim mechanism makes it possible to reduce the stick force to zero regardless of the stick (or control surface) position. Figure II-18 (b) gives the stick force equation of the simple spring, showing that the stick force is directly proportional to the control surface deflection; Figure II-18 (c) illustrates this relation. Figure II-18 (d) shows the block diagram representing this simple spring system.



$$\Delta\delta = \frac{1}{K} F$$

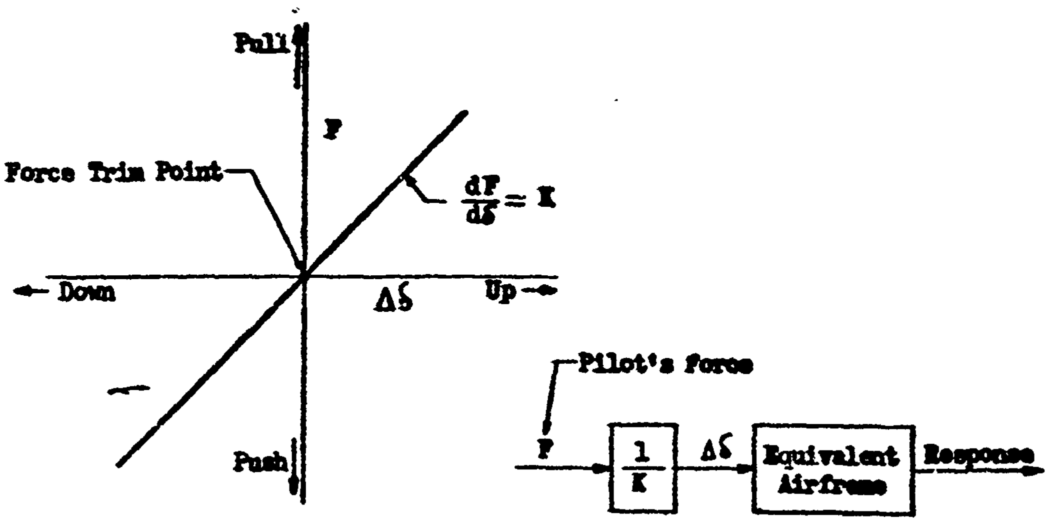
F = Stick force, lb

K = Mechanical spring constant (including stick-to-control surface gearing), lb/deg

$\Delta\delta$ = Control surface deflection from force trim position, deg

(a) Schematic

(b) Equation



(c) Characteristics

(d) Block Diagram

Figure II-18. Simple Spring in Elevator Control System

Based on the typical responses of high-speed airplanes as presented in the preceding section, Figure II-19 shows typical control feel responses of an airplane using a simple spring in the artificial feel system.

It is seen that the longitudinal stick-free static stability as given by the slope of F_E versus Mach number in Figure II-19(a) is rather poor. The gradient is too high at low speeds, indicating a high landing stick force, and the reversal region near the transonic Mach numbers is undesirable. The curves of Figure II-19(b) show a very large variation of stick force per g over the flight egime of the airplane, and also show large altitude and Mach number effects all of which are undesirable. Figure II-19(c) shows an appreciable decrease in lateral force feel for supersonic Mach numbers, and it also shows a large altitude effect. Figure II-19(d) shows that the directional feel is acceptable because there is only a relatively small increase of $F_A/(pb/2V)$ as Mach number increases and because there is practically no effect of altitude on directional feel.

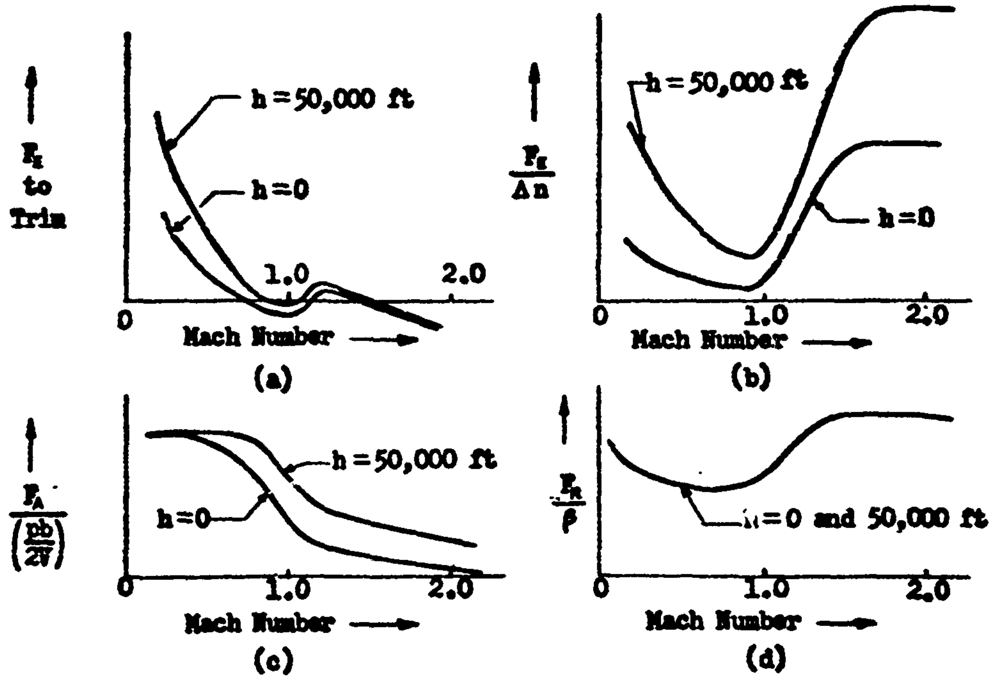


Figure II-19. Control Feel Curves for a Typical High-Speed Airplane with a Simple Spring Only

In summary, an artificial feel system using a spring only, would probably exhibit very poor longitudinal and lateral control feel characteristics for a typical supersonic airplane. However, the directional control feel characteristics for such an airplane might be acceptable.

(b) PRELOADED SPRING

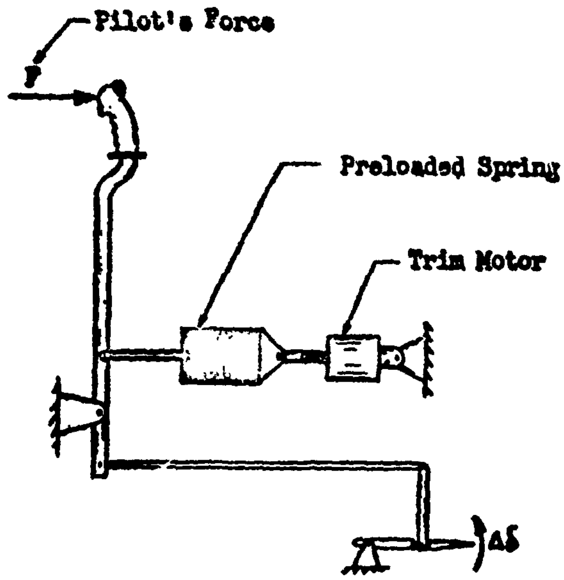
The presence of friction in a control system prevents the control stick and/or the control surface from returning to a trim position when external forces are removed; i.e., friction creates poor stick centering tendencies in a control system. One of the most common purposes of a preloaded spring is to improve the stick centering characteristics of a simple spring artificial feel system.

A schematic and the characteristics of a preloaded spring system are shown in Figure II-20. The initial steep gradient of the F versus $\Delta\delta$ curve shown in Figure II-20(c) indicates that a large force is required to break away from trim position. If this force is of the same magnitude as the friction force, good self-centering characteristics can be assured.

Additional nonlinearities in the F versus $\Delta\delta$ curve can be created by using several springs preloaded at different values. Figure II-21 shows a double preloaded spring system. The characteristics shown in Figure II-21(c) may be desirable in some control systems. These characteristics are

1. High initial gradient for good self-centering characteristics,
2. Moderate gradient in the intermediate, or maneuvering, range to lessen pilot fatigue, and
3. High gradient at extreme deflections to act as a warning to pilot.

Because of the nonlinear force gradients in preloaded spring systems, it is difficult to define control feel characteristics in terms of our existing criteria as was done for the case of the simple spring. A practical solution to this problem is to approximate the nonlinear force gradients with an average



(a) Schematic

$\Delta\delta = 0 \quad \text{for } F < P$

$\quad = \frac{F-P}{K} \text{ for } F > P$

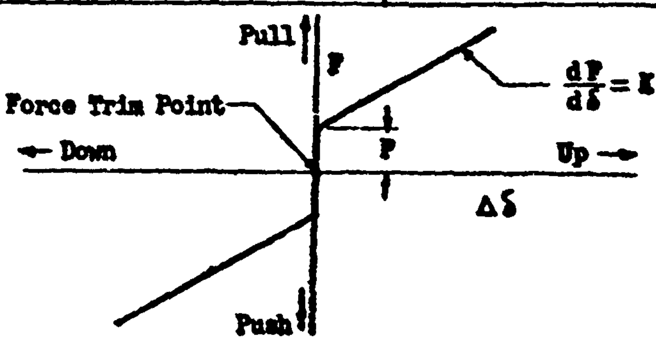
F = Stick force, lb

P = Preload force, lb (P takes the same sign as F)

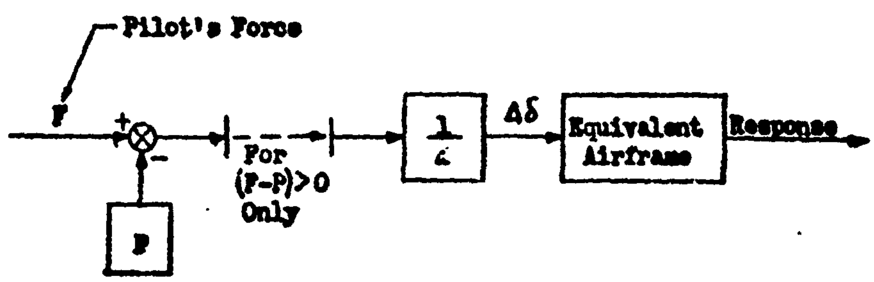
K = Spring constant (including stick-to-control surface gearing) lb/deg

$\Delta\delta$ = Control surface deflection from force trim position, deg

(b) Equation

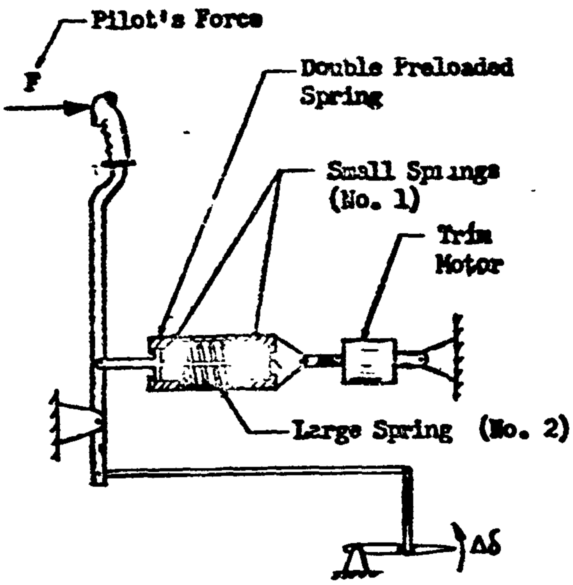


(c) Characteristics

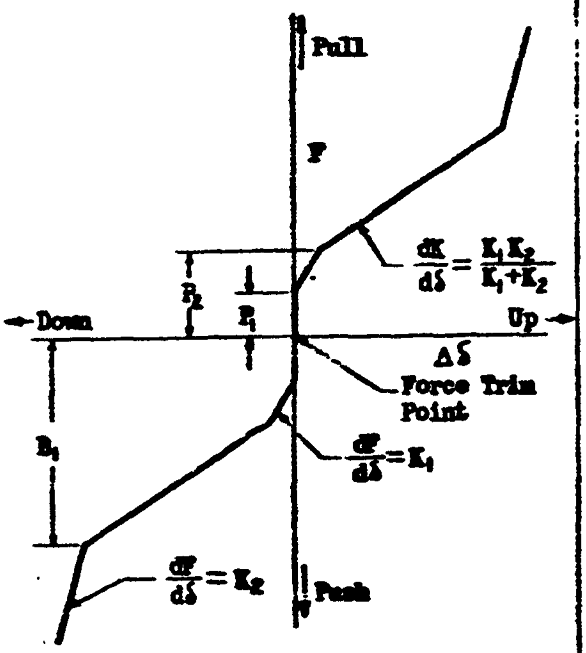


(d) Block Diagram

Figure II-20. Preloaded Spring in Elevator Control System



(a) Schematic



(c) Characteristics

$$\Delta\delta = 0 \quad \text{for} \quad F < P_1$$

$$= \frac{F-P_1}{K_1} \quad \text{for} \quad P_1 < F < P_2$$

$$= \frac{F - \dfrac{P_1K_2+P_2K_1}{K_1+K_2}}{K_1K_2/K_1+K_2} \quad \text{for} \quad P_2 < F < B_1$$

$$= \frac{F - \left[P_2 + \dfrac{K_2}{K_1}(P_1 - B_1)\right]}{K_2} \quad \text{for} \quad B_1 < F$$

F = Stick force, lb

P_1 = Preload force of small springs, lb

K_1 = Spring constant of small springs (including stick-to-control surface gearing), lb/deg

B_1 = Bottoming force of small spring, lb

P_2 = Preload force of large spring, lb

K_2 = Spring constant of large spring (including stick-to-control surface gearing) lb/deg

Note: The signs of P_1, P_2, and B_1 take the same sign as F.

(b) Equations

Figure II-21. Double Preloaded Spring in Elevator Control System

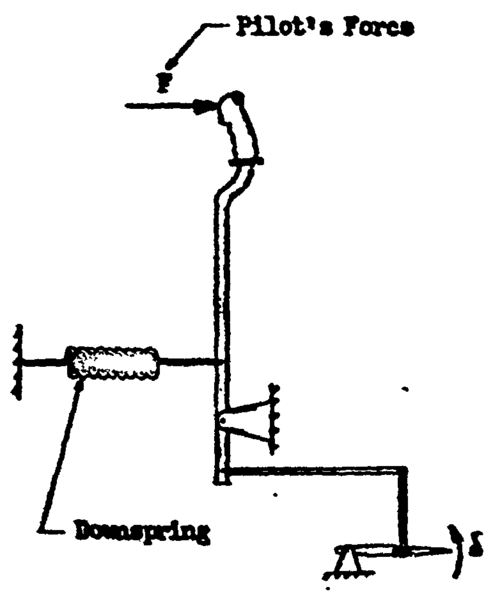
linear gradient and then to use the existing control feel criteria. Using this procedure, the control feel curves for a preloaded spring system would be similar to those shown in Figure II-19. For a more detailed control feel analysis of a preloaded spring system, the particular airplane and the particular non-linear force characteristics of the feel system must be known.

(c) DOWNSPRING

The downspring is used to improve the longitudinal stick-free static stability, i.e., the stick force to trim characteristics of an airplane, by effectively increasing the gradient of the curve of force versus Mach number. The device consists of a preloaded spring which has a low spring constant and which is attached to the control stick so as to produce an approximately constant pull force which is independent of the speed of the airplane. The pilot experiences a force from the downspring as indicated in Figure II-22.

The preload, P, is usually large in comparison with the product $K\delta$ in order to prevent excessively high stick forces at low speeds where large up-elevator deflections are required. Notice that δ is to be distinguished from $\Delta\delta$ as used elsewhere. Here δ refers to elevator deflection from a neutral position whereas $\Delta\delta$ refers to an elevator deflection from a stick force trim position.

The downspring primarily affects the stick force to trim curve, and it acts to increase the gradient, as shown in Figure II-23. One objection to the use of a downspring is that a heavy, unnatural pull force is required to hold the stick back during taxiing, take-off, and landing operations of the airplane.



(a) Schematic

$$\delta = \frac{1}{K}(F-P)$$

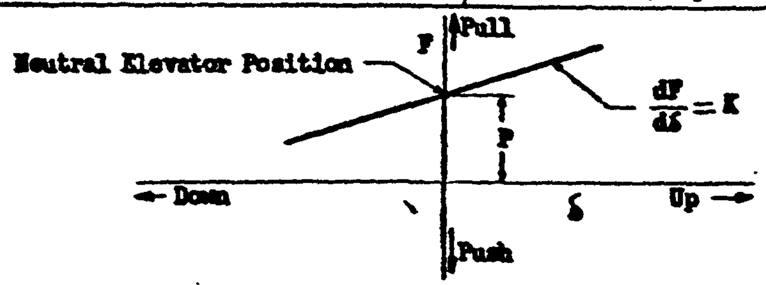
F = Stick force, lb

P = Downspring preload at neutral elevator deflection, lb

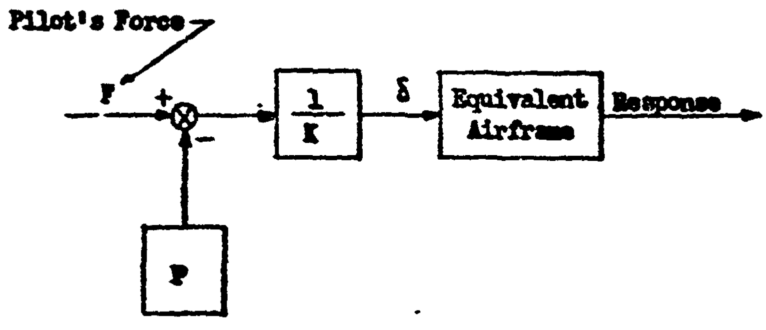
K = Downspring constant lb/deg

δ = Elevator deflection from neutral, deg

(b) Equation



(c) Characteristics



(d) Block Diagram

Figure II-22. Downspring in Elevator Control System

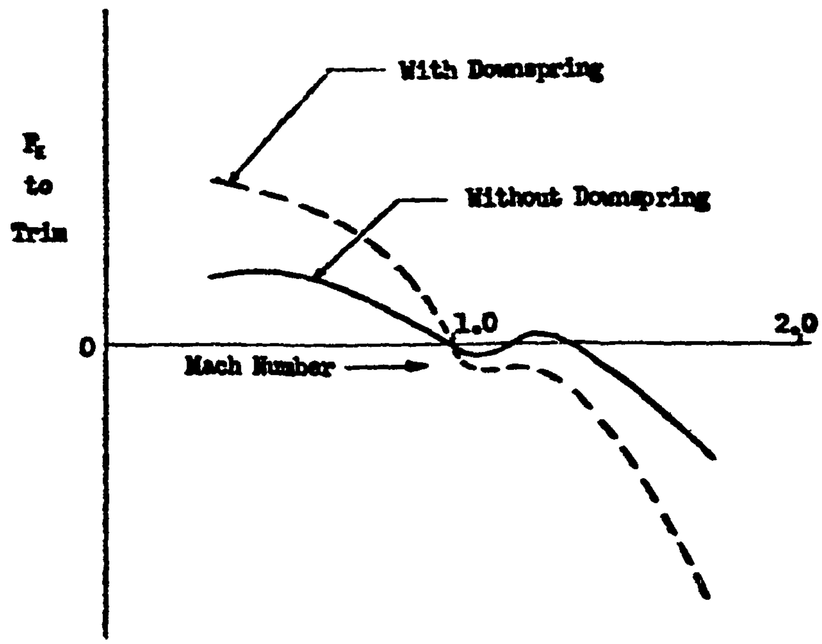


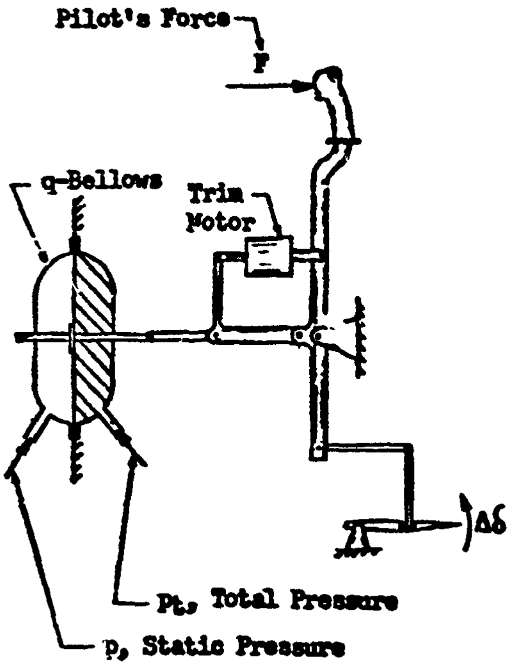
Figure II-23. Typical Effect of Downspring on Stick Force to Trim versus Mach Number.

(d) q-BELLOWS

One method of improving the control feel characteristics with a rather simple mechanical feel system is to use a q-bellows. Instead of a spring gradient that is constant throughout the flight regime of the airplane, the q-bellows provides a variable spring gradient that is a function of Mach number and altitude. Thus the q-bellows can be thought of as a mechanical gain changer, or gain compensator.

A typical q-bellows system, as shown in Figure II-24, produces a stick force proportional to the product of the pressure differential across the diaphragm of the bellows, $p_t - p$, and the control surface deflection. The pressure differential, $p_t - p$, can be conveniently expressed in terms of dynamic pressure, q, as

$$p_t - p = \left(\frac{p_t - p}{q}\right) q$$



(a) Schematic

$$\Delta\delta = \frac{F}{K(p_t - p)}$$

F = Stick force, lb

K = Gearing constant, stick-to-control surface ft^2/deg

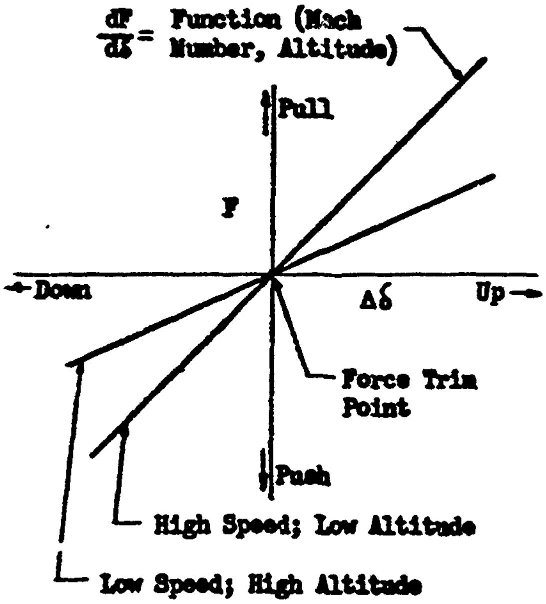
(p_t-p) = Pressure differential across bellows, lb/ft^2

p_t = Total pressure, lb/ft^2

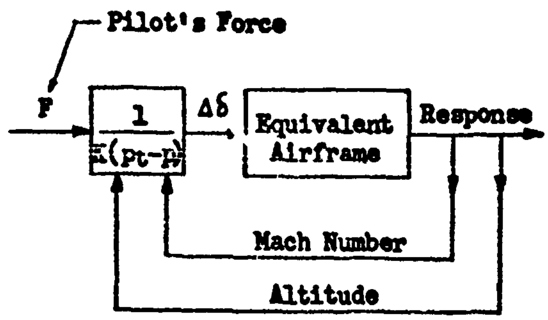
p = Static pressure, lb/ft^2

$\Delta\delta$ = Control surface deflection from force trim position, deg

(b) Equation



(c) Characteristics



(d) Block Diagram

Figure II-24. The q-Bellows in Elevator Control System

Dynamic pressure is a function of the speed and altitude at which the airplane is flying and is given by either of the following relations:

$$q = \frac{1}{2}\rho U^2 = .7 p M^2$$

where

q is the dynamic pressure, lb/ft^2

ρ is the ambient air density, slugs/ft^3

U is the true airspeed, ft/sec

p is the static pressure, lb/ft^2

M is the Mach number

The ratio $(p_t - p)/q$ is primarily a function of Mach number. However, in any practical bellows application, the pressure sensing device is almost always located within the pressure field around the airplane, in which case the pressure ratio $(p_t - p)/q$ may be a function of other variables, such as angle of attack and sideslip angle. Assuming, however, that the pressure sensing device is a conventional pitot tube, and neglecting any interference caused by the presence of the airplane, the ratio $(p_t - p)/q$ is then a function of Mach number only, as shown in Figure II-25. For supersonic Mach numbers, the values shown in Figure II-25 include the pressure loss through the normal shock wave ahead of the pitot tube.

Based on the same typical responses presented for the simple spring, the typical control feel responses of a q-bellows system shown in Figure II-26 can be expected.

Figure II-26(b) shows that the use of a q-bellows improves the stick force per g characteristics for low Mach numbers by bringing the altitude curves together and by reducing the variation of $F_s/\Delta n$ with Mach number for very

low Mach numbers; however, for transonic and supersonic Mach numbers, the stick force gradient is too high. One method of decreasing the stick force gradient at high Mach numbers is to provide a bellows pressure relief or cut-off for high Mach numbers. Figure II-26(c) shows a more constant trend of $F_r/(pb/2V)$ with Mach number than did the simple spring system, but Figure II-26(d) shows that the directional feel characteristics vary with Mach number and altitude in a q-bellows system. However, for structural reasons (vertical tail loads), the trend of higher rudder forces for higher Mach numbers may be very desirable.

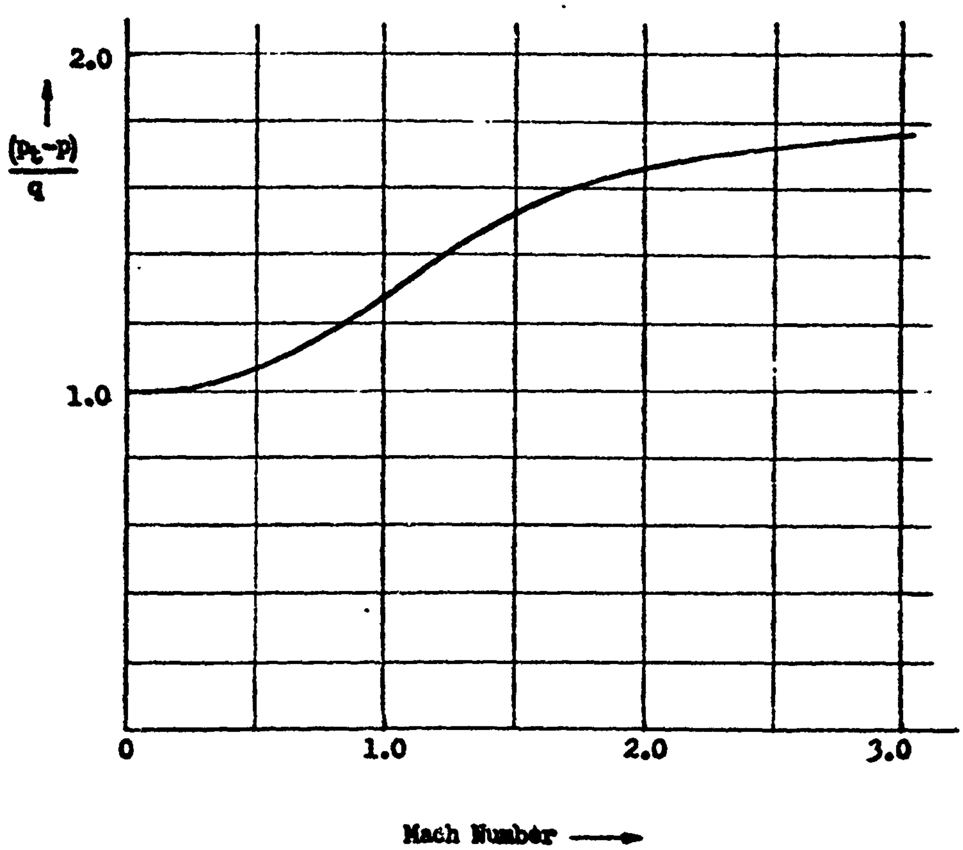


Figure II-25. The Ratio of Pressure Differential to Dynamic Pressure, $(p_t - p)/q$, versus Mach Number

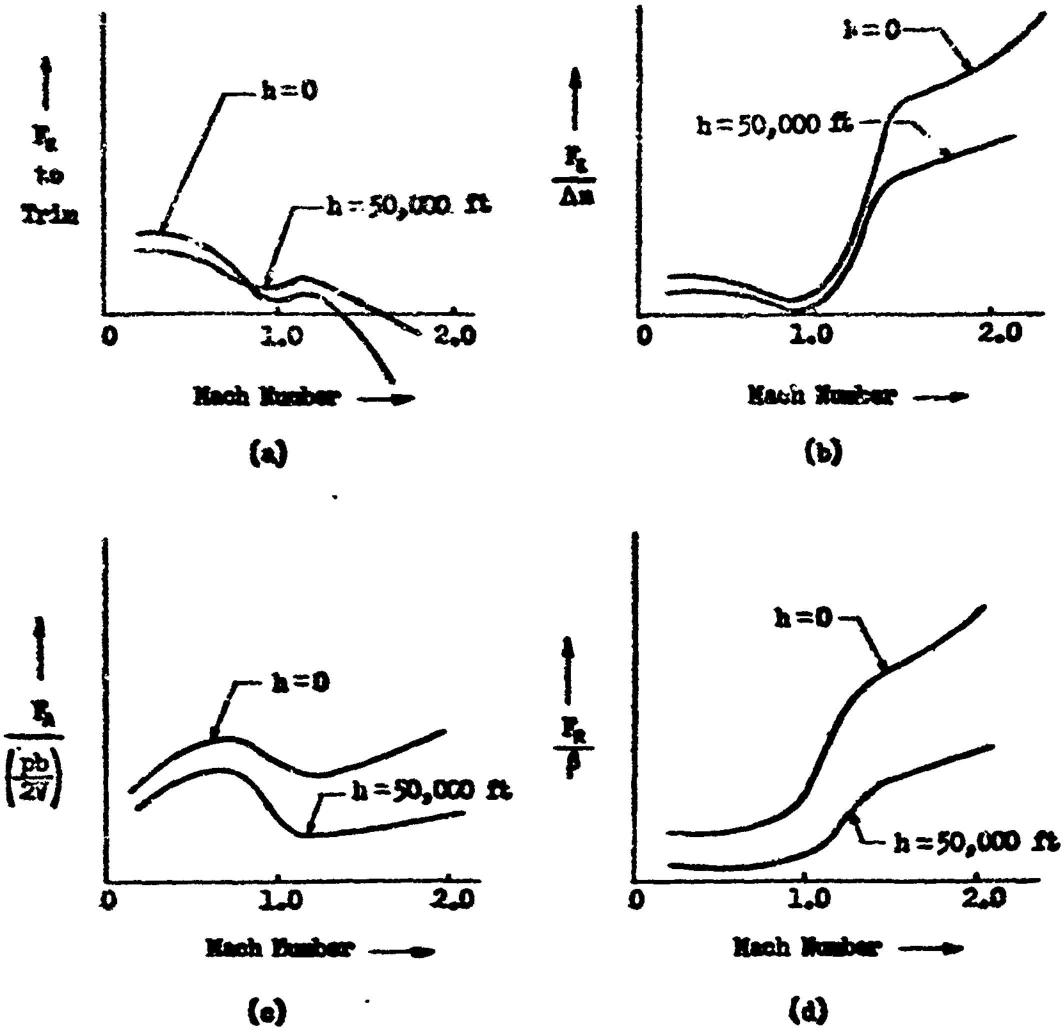


Figure II-26. Control Feel Curves for a Typical High-Speed Airplane with a q-Bellows Only

The foregoing discussion has shown that a typical airplane with an artificial control feel system using only a q-bellows would probably exhibit acceptable lateral and directional feel characteristics; it would probably exhibit acceptable longitudinal feel characteristics if suitable bellows relief were provided at supersonic Mach numbers.

(e) THE RATIO CHANGER

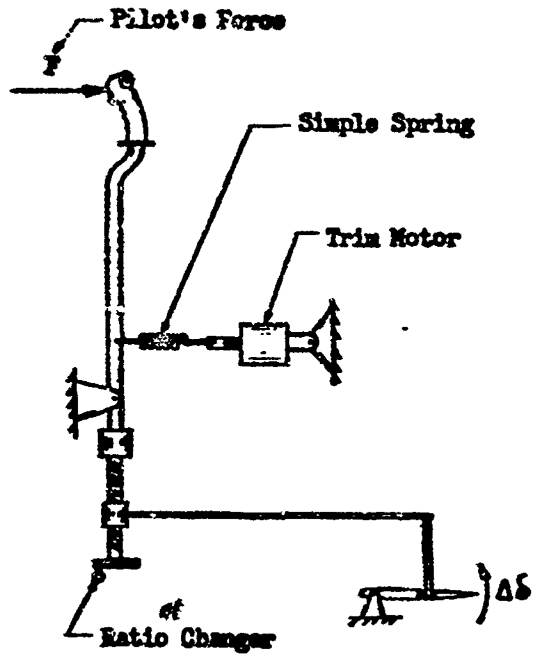
The ratio changer is a mechanical device for providing a variation of the stick-to-elevator gearing constant as a function of Mach number, altitude, or possibly c.g. position; in other words, it is a gain changer. Whereas the q-bellows provides "automatic" gain compensation as a function of the flight conditions, the ratio changer must be positioned by external forces.

The source of power for the ratio changer can be either the pilot or a servo positioning system which uses a Machmeter and/or an altimeter for sensing. The servo-positioned ratio changer requires no effort on the part of the pilot; however, this feature can be achieved only at the expense of lowered system reliability. The optimum system as conceived at present would probably be servo-positioned with provisions for a manual over-ride in case of a servo failure.

It is possible to incorporate the ratio changer into any type of feel system; however, it is usually considered only in conjunction with a simple or pre-loaded spring system. Figure II-27 illustrates the application of a ratio changer to a simple spring system.

Usually the ratio changer is used in the longitudinal system only, primarily to provide a stick force per g which is constant with Mach number and altitude. Figure II-28 shows, however, that if the $F_s/\Delta n$ curve is made constant with Mach number and altitude, the stick force to trim curves may show a more severe unstable slope in the tuck region. This effect, however, depends on the particular airplane, and in some cases the application of a ratio changer may greatly improve both $F_s/\Delta n$ and F_s to trim curves.

In cases of particularly undesirable lateral control feel characteristics, a ratio changer may be of benefit in the aileron control system.



(a) Schematic

$$\Delta\delta = \frac{F}{RK}$$

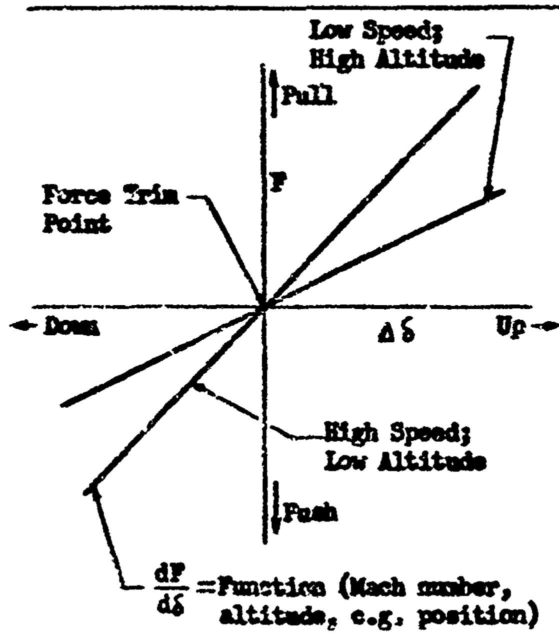
F = Stick force, lb

R = Ratio changer stick-to-control surface gearing (variable as a function of Mach number, altitude, and c.g. position), dimensionless

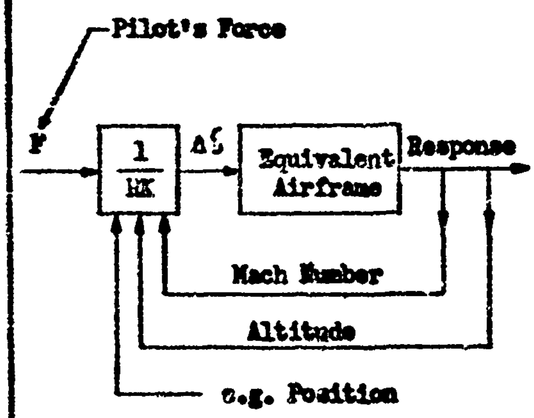
K = Spring constant (fixed), lb/deg

Δδ = Control surface deflection from force trim position, deg.

(b) Equation



(c) Characteristics



(d) Block Diagram

Figure II-27. Ratio Changer and Simple Spring in Elevator Control System

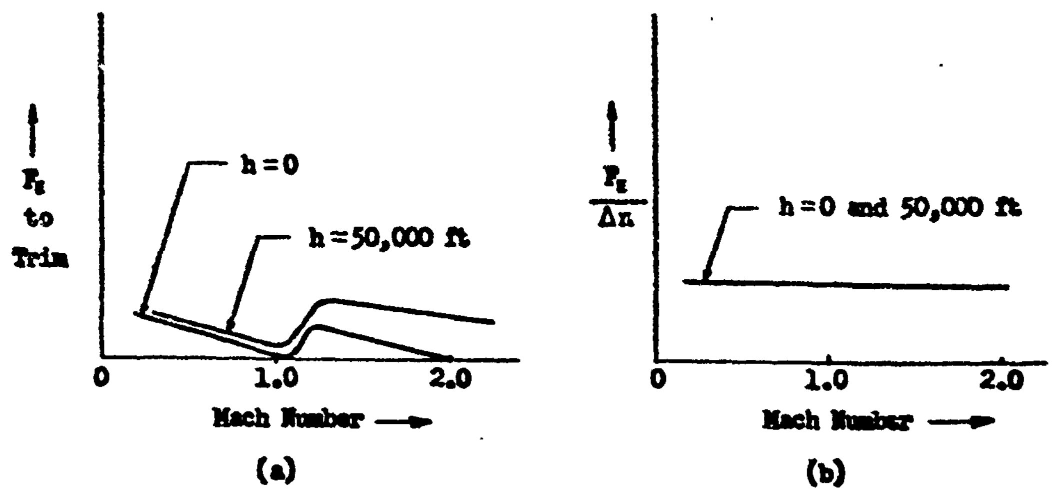


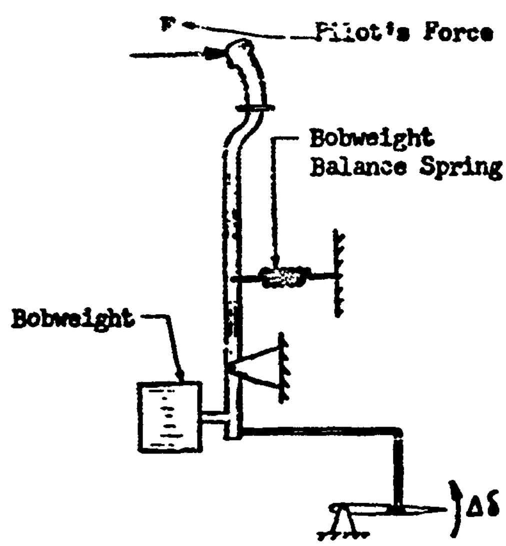
Figure II-28. Control Feel Curves for a Typical High-Speed Airplane with a Ratio Changer and Simple Spring

(f) BOBWEIGHT

The bobweight is a simple mechanical device which provides a stick force proportional to normal acceleration and thus improves maneuvering feel. It consists of a weight mounted on the control stick in such a way that it tends to cause a forward movement of the stick. The more normal acceleration the airplane is subjected to, the more the stick will tend to move forward. If the pilot resists this movement, he will feel a stick force directly proportional to the increment of normal acceleration.

Figure II-29 shows a typical schematic of a bobweight installation and some of its characteristics. In many applications, the undesirable stick force created by the bobweight when the stick is in a neutral position is canceled by a bobweight balance spring.

Figure II-30 illustrates a typical control feel response with and without a bobweight. It is seen that the bobweight merely increases the stick force



(a) Schematic

$F_s = K\Delta n$

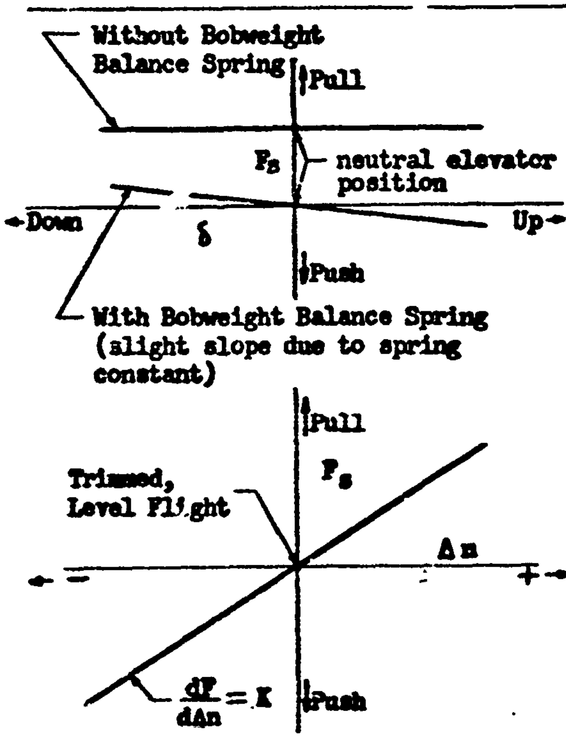
F_s = Stick force, lb

K = Bobweight effectiveness, lb/g

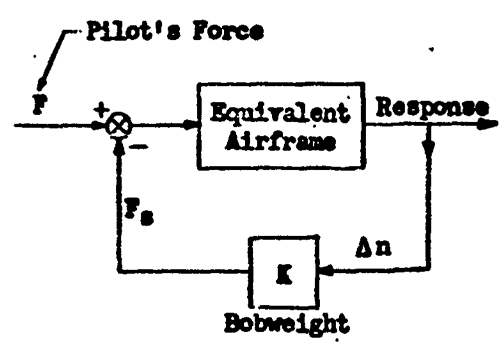
Δn Increment of normal acceleration, g's ($\Delta n = n-1$)

Note: This equation assumes the bobweight balance spring constant is negligibly small.

(b) Equation



(c) Characteristics



(d) Block Diagram

Figure II-29. Bobweight in Elevator Control System

per g by a constant amount. If the bobweight balance spring is not used, the addition of a bobweight to a longitudinal control feel system will also change the F_s to trim curve. For this case, the bobweight acts like a downspring and tends to produce a more stable gradient as was shown in Figure II-29.

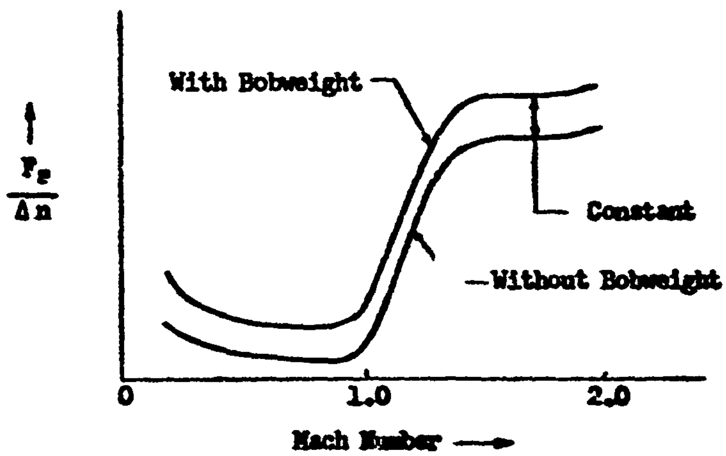


Figure II-30. Effect of Bobweight on Stick Force per g

There is one great objection to the use of a bobweight in high speed airplanes: it may cause poor transient feel because of the lag between normal acceleration response and command input.

A precaution that must be taken when designing a bobweight control feel system is to eliminate the possibility of coupling between the bobweight and airplane natural frequencies. For flight at high speeds, where the longitudinal short period frequency is very high, there is a definite possibility of such a coupling effect which could result in uncomfortable (and perhaps dangerous) pitching oscillations in gusty weather flight. These points make it clear that if a bobweight is to be included in a control feel system, the dynamic characteristics of the integrated system must be carefully studied.

(g) DAMPER

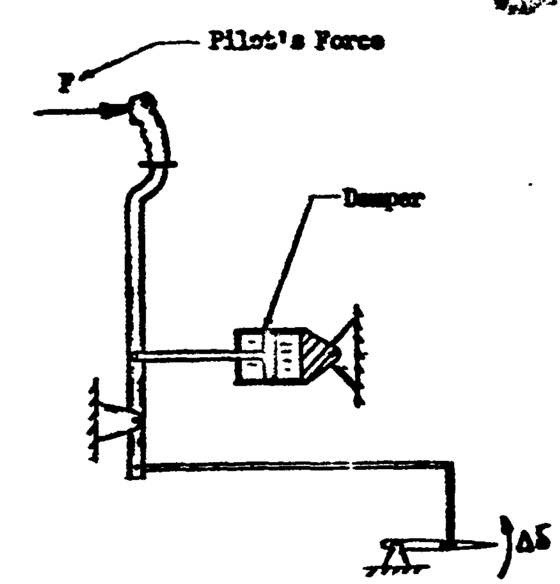
The purpose of the damper is to provide a stick force proportional to the rate of stick deflection. Mechanically this device consists of a small piston moving within a cylinder of oil, the motion of the piston being restricted by oil which must be forced through tiny orifices in the piston. When the pilot deflects the stick, he will experience a force proportional to stick (or elevator) velocity. A schematic and the characteristics of a damper system are shown in Figure II-31.

The damper is used in longitudinal control feel systems to improve the transient feel if an airplane exhibits unsatisfactory transient feel characteristics. The effect of a damper can be seen by referring to Figure II-8, where the curves in Figure II-8(a) are for a control feel system which has a damper, and the curves in Figure II-8(b) are for a system without a damper.

The damper appears to be a satisfactory solution to the transient feel problem, but it has several drawbacks. One is that if the damper effectiveness, K, is selected for some critical flight condition where the need for the damper is greatest—usually in the transonic region—it will probably be found that the damper restricts the maneuverability at other flight conditions where high rates of elevator motions are necessary. In other words, in order to make a damper operate successfully, the damper effectiveness must be made a function of Mach number, altitude, and possibly c.g. position. Another drawback is the difficulty in designing a damper which can operate at various temperature levels and which will provide sufficient damping for small stick deflection.

(h) SERVO SYSTEMS

Most of the purely mechanical control feel devices described previously have the common failing that their effectiveness must be a function of flight condition and airplane configuration.



(a) Schematic

$$\Delta\dot{\delta} = \frac{F}{K}$$

$$\Delta\delta = \frac{F}{Ks}$$

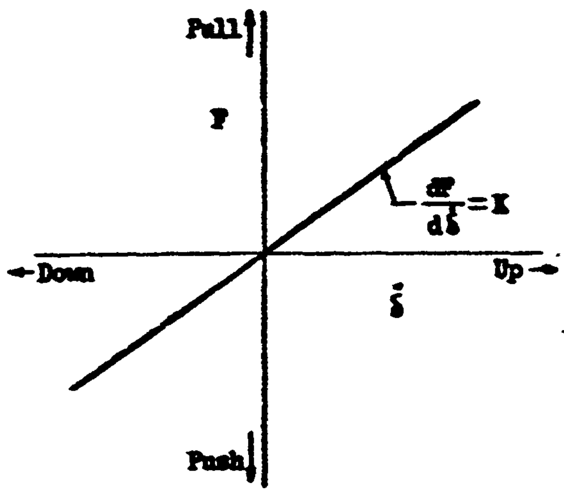
F = Stick force, lb

K = Damper effectiveness, $\frac{\text{lb}}{\text{deg/sec}}$

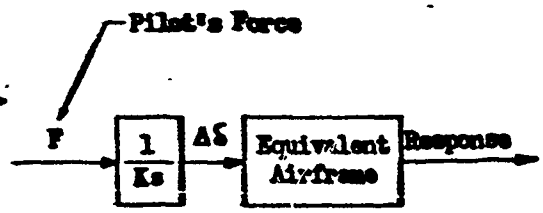
$\Delta\dot{\delta}$ = Stick (or elevator) velocity, deg/sec

s = Laplace transform (essentially equal to d/dt)

(b) Equation



(c) Characteristics



(d) Block Diagram

Figure II-31. Stick Damper in Elevator Control System

Most present-day feel systems incorporating only these mechanical devices without adequate "gain" compensation as a function of Mach number, altitude, and c.g. position have not been entirely satisfactory. It is unreasonable to expect the pilot himself to provide the necessary gain functions because his attention may be required elsewhere. For instance, the pilot certainly could not be expected to provide compensation to the feel system during combat maneuvering flight where extremely wide Mach number and altitude changes occur.

To provide satisfactory feel characteristics throughout the flight regime of a supersonic airplane, it is becoming more and more apparent that servo systems incorporating automatic gain compensation will be necessary. There are two lines of approach in designing servo systems into an airplane. First, suitable motion stability augmenting servo systems can be added to the basic airframe block, thus creating desirable airframe responses to pilot's force inputs. An entirely mechanical type feel system may then be adequate to provide satisfactory control feel. On the other hand, if the basic airframe is not suitably augmented, force augmentation in the feel system can be provided by a servo device. In many cases, because of the complexity of the problem, servo devices with automatic gain compensation will probably be needed for both motion and force stability augmentation.

There are many ramifications of the servo type artificial feel producer with automatic gain compensation. Some are of the open center hydraulic type with Mach number and altitude compensation in the form of flow restrictors. Others are of the electrical slipping clutch type with appropriate monitoring from Mach and altitude sensors. Most of these devices are specifically designed to meet the requirements of a given airplane.

1. "Flying Qualities of Piloted Airplanes," U.S. Air Force Specification No. R-18[illegible]B, June 1, 1948.

2. "Specification for Flying Qualities of Piloted Airplanes," Bureau of Aeronautics Specification. NAVAER SR-119B, June 1, 1948.

3. Perkins, Courtland D., and Hage, Robert E., Airplane Performance Stability and Control, John Wiley & Sons. New York, 1949.

4. Second Piloted Aircraft Flight Control System Symposium, Parts I-V, BuAer Report No. AE-61-5, Airborne Equipment Division, Bureau of Aeronautics, Department of Navy, Washington, D. C. Symposium held June 2, 3, 4, and 5, 1952. (Confidential)

5. A Compilation of Papers Presented, NACA Conference on High-Speed Airplane Aerodynamics, Langley Aeronautical Laboratory, Langley Field, Virginia. Conference held December 4-5, 1951. (Confidential)

CHAPTER III

DESIGN PROCEDURE

SECTION 1 - INTRODUCTION

In Chapters I and II, the separate elements of the pilot-airframe system were briefly discussed. These elements were:

1. The controlled element: the basic airframe.
2. The controlling elements:
    a. Sensing and actuating element: the human pilot.
    b. Equalization: the artificial feel system.

The artificial feel system is the equalization for the complete system in accordance with the definitions given in Chapter IV, Section 1(a) of Reference 1. First, the artificial feel system is almost completely alterable within the bounds of physical realizability. Second, this equalization is used to tie the unalterable controlled and controlling elements into a well-integrated functional system with specified system requirements.

The purpose of this chapter is to outline a method for designing the artificial feel system. It is important to note that this method is equivalent to the one used by the various agencies when the specifications for flying qualities were established. The latter method involved performing extensive flight tests of many airplanes; the method to be used in this chapter accomplishes the same task through simulation of the pilot-airframe system on the ground.

Section 2 presents the basic concepts behind this design philosophy. Section 3 details a program for obtaining a solution to the problem by means of the analog computer. In addition, some of the quantitative and qualitative results from such a program are included. Section 4 is devoted to a brief discussion of the physical mechanization of the results obtained in Section 3.

SECTION 2 – BASIC CONCEPTS

(a) GENERAL CONSIDERATIONS

The basic pilot-airframe system has the same general form as any servo system. This form is shown in Figure III-1.

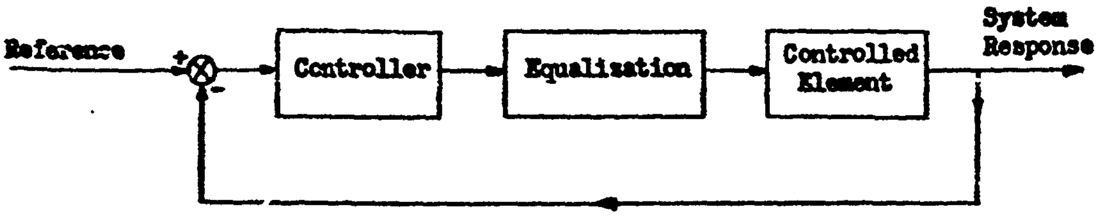
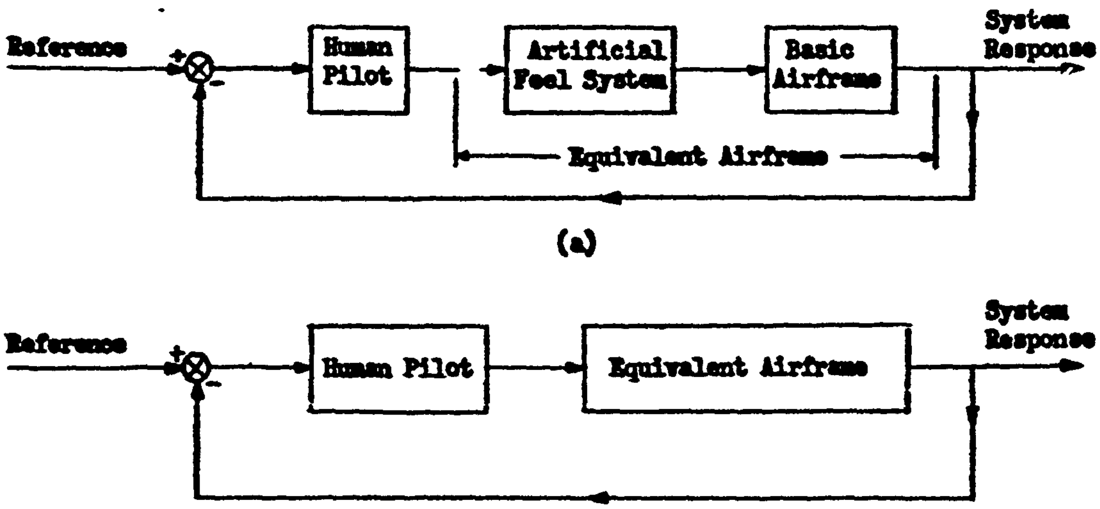


Figure III-1. Functional Block Diagram of Basic Servo System

Replacing the functional blocks by their physical counterparts gives the block diagram for the pilot-airframe system, Figure III-2:



(b)

Figure III-2. Functional Block Diagram of Basic Pilot-Equivalent Airframe System

Figure III-2 shows the artificial feel system and the basic airframe lumped together as one block. This procedure will be justified later in the chapter,

and it wi... further shown th... the ... solution to the ... problem is greatly s...

(b) THE CONTROLLER

Consider now the longitudinal dynamics of the system. What are the inputs to the pilot? The only stimuli that the pilot can use for flight control are either visual ones or those resulting from the dynamic forces acting on his body. Of the two, the visual input is more useful as an aid in directing the flight path.

The visual input can come from several sources, such as the position of the horizon or the indications on the cockpit instruments. Assume that the pilot's visual input comes from the position of the horizon and is transmitted as the difference between the desired pitch attitude θ_{ref} and the actual pitch attitude θ.

Corresponding to this input is the pilot's output, an applied force F_p on the cockpit controls. The block diagram of Figure III-2 is then revised as shown in Figure III-3.

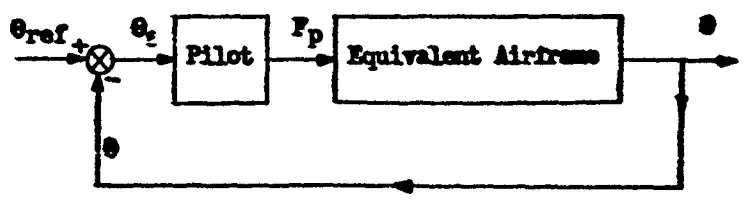


Figure III-3. Pilot-Equivalent Airframe System Block Diagram with Pilot Input

The pilot applied force F_p is given by

(III-1) $F_p = Y_p \theta_\epsilon$

where Y_p is the pilot's transfer function relating the force output to the visual input.

Much experimental work has been and is being done to obtain suitable forms and values for the pilot transfer function. For the work to follow, the transfer function developed by the Goodyear Aircraft Corporation is used (Reference 3). With respect to the form, this transfer function has been found to be reasonably acceptable. This form is

$$(III\text{-}2) \qquad Y_p = K e^{-\tau s}\left(\frac{T_1 s + 1}{T_2 s + 1}\right)$$

The gain term is largely a function of experience, fatigue, and tenseness. Under normal conditions, the pilot will adjust his gain to best suit the rest of the system.

The factor $e^{-\tau s}$ determines the fixed dead-time between the pilot's response and his input stimulus (pilot's reaction time).

The denominator factor $T_2 s + 1$ is a measure of the pilot's neuromuscular lag. That is, there is a lag of $3T_2$ seconds between the time the pilot's response is initiated and approximately 95% completed. The effects of the fixed dead-time, or reaction time delay, and the lag are shown in Figure III-4 for a step function stimulus.

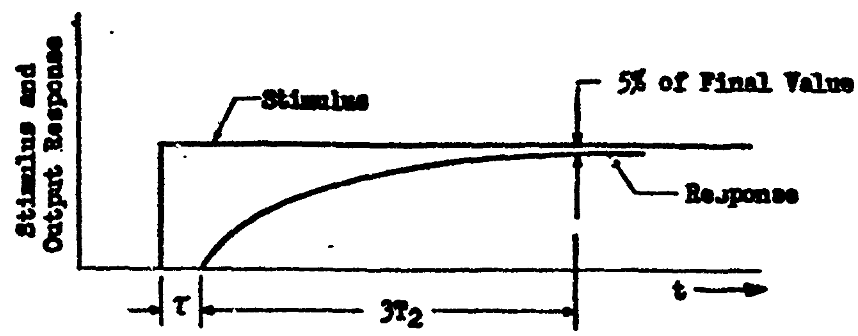


Figure III-4. Reaction Time and Neuromuscular Lag Effects

The numerator term $T_1 s + 1$ is a function of the pilot's rate judgment. The significance of this term can be viewed in the following manner.

Through experience, the pilot knows that he is controlling an element, the airframe, which has a lag inherently associated with it; that is, the airframe output motion will lag any command input that the pilot transmits. To offset this lag, the pilot subconsciously controls the rate of his input command so as to decrease the effective airframe lag as best he can. This action is illustrated in the idealized Bode diagram of Figure III-5.

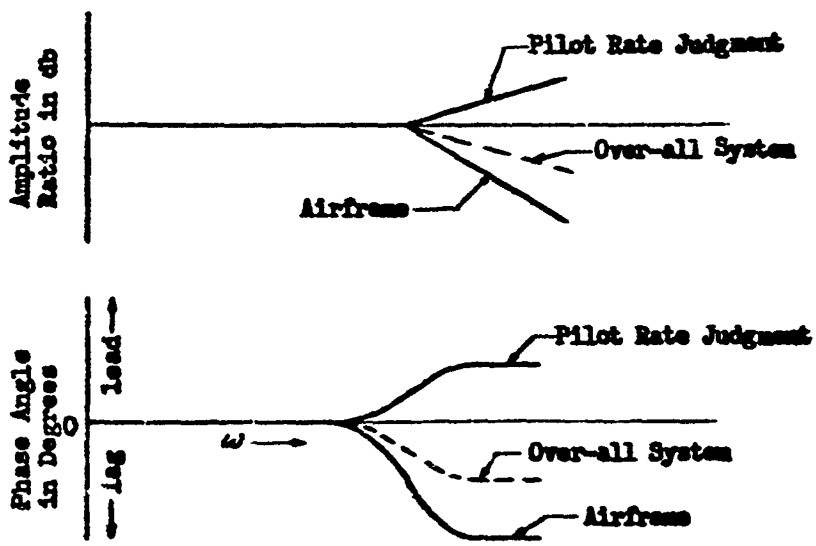


Figure III-5. Effects of Pilot Rate Judgment on Airframe Response

Figure III-5 indicates that the pilot rate judgment term decreases the apparent airframe lag, thus improving the system. In effect, the $T_1 s + 1$ term acts as an equalizer in the complete closed loop system.

In addition to the terms discussed above, the pilot transfer function, (III-2), should include a factor simulating threshold effects. There are two types of thresholds to be considered in simulating the pilot. First, there

is the perceptual threshold; that is, there are certain values of stimulus below which the pilot cannot sense any input. Above these values the pil[illegible] sense the stimulus and respond accordingly. This is shown in Figures III-6(a) and III-6(b). The effects of this type of threshold can be minimized by magnifying the presentation on the cockpit indicators.

The second and possibly more important threshold is the "indifference threshold." This is shown in Figures III-6(c) and III-6(d). For any stimulus within the threshold range, the pilot simply does not care and does nothing. When the stimulus exceeds the threshold value, the pilot then responds as if there were no threshold. It is important to note that the indifference threshold occurs at the pilot's output while the perceptual threshold is an input phenomenon.

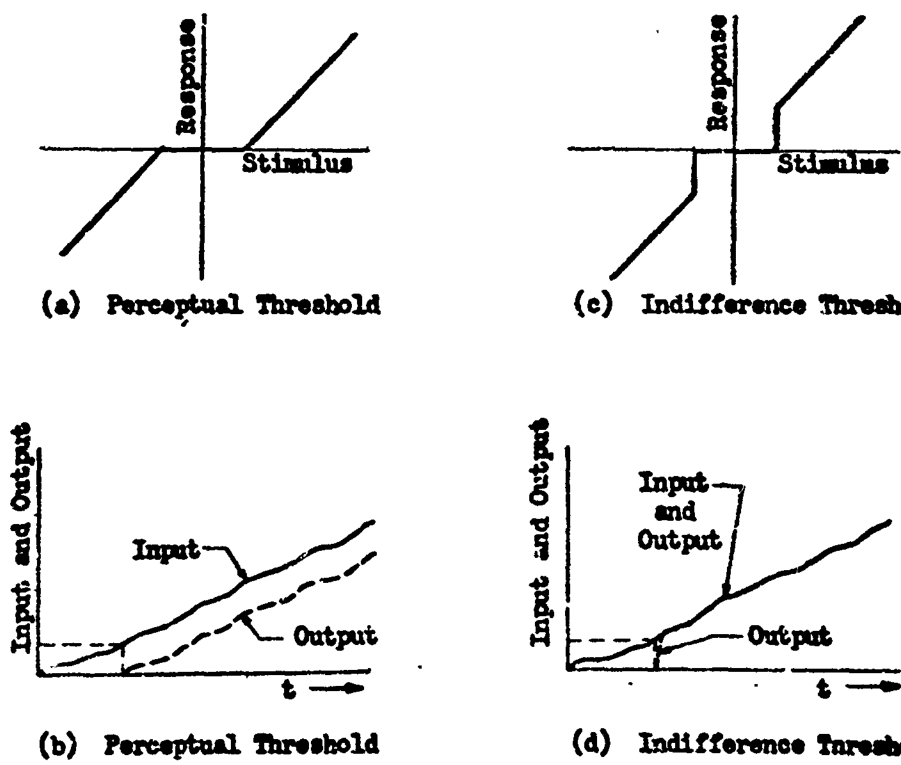


Figure III-6. Pilot's Threshold Effects

It is understood that (III-2) may not give a true representation of the human pilot, in which case the results presented in the following analysis will be in error. If the human pilot transfer function were known exactly, any analysis using the exact transfer function would give correct results. However, the main objective of this chapter is to present the concepts behind, and an insight into, the analytical investigation of pilot-airframe systems. Until the time when the exact human pilot transfer function is known, (III-2) will serve as a guide for future studies.

(c) THE CONTROLLED ELEMENT

The equations of longitudinal motion of the basic airframe are: *

$$
\text{(III-3)}\quad \begin{cases} \dot{u} = X_u u + X_w w - g\theta & \text{Forward Acceleration Equation} \\ \dot{w} - U_0\dot{\theta} = a_z = Z_u u + Z_w w + Z_{\dot{\theta}}\dot{\theta} + Z_{\delta_E}\delta_E & \text{Normal Acceleration Equation} \\ \ddot{\theta} = M_u u + M_w w + M_{\dot{w}}\dot{w} + M_{\dot{\theta}}\dot{\theta} + M_{\delta_E}\delta_E & \text{Pitching Acceleration Equation} \end{cases}
$$

Equations (III-3) assume that the elevator is the only control available for longitudinal dynamics. This assumption will be carried throughout the following analysis, and the effects of speed brakes, flaps, and throttle will be disregarded.

Furthermore, (III-3) shows that the elevator affects only the normal and pitching acceleration equations. Consequently, any augmentation provided by the equaliser through the elevator can affect only the Z and M, or normal force and pitching moment stability derivatives.

Equations (III-3) yield the characteristic equation for the airframe for elevator deflections. This equation is of the form **

$$
\text{(III-4)}\qquad \Delta = As^4 + Bs^3 + Cs^2 + Ds + E
$$

* See Reference 4, Equati... (III-9) and (III-24).
** See Reference 4, Table I... III-26.

where

$$A = 1$$

$$B \approx -\left[U_0 M_{\dot{w}} + M_{\dot{\theta}} + Z_w\right]$$

$$C \approx M_{\dot{\theta}} Z_w - U_0 M_w$$

$$D \approx -X_u(M_{\dot{\theta}} Z_w - U_0 M_w) - M_u(X_w U_0 - g)$$

$$E \approx g(Z_u M_w - M_u Z_w)$$

The relative magnitudes of the coefficients are such that

$$\text{(III-5)*} \quad \Delta \approx (s^2 + Bs + C)\left(s^2 + \frac{DC - BE}{C^2} s + \frac{E}{C}\right)$$

The first and second factors in (III-5) describe the characteristic longitudinal short period and phugoid modes of the airframe respectively. Equation (III-5) is then rewritten as

$$\text{(III-6)} \quad \Delta \approx (s^2 + 2\zeta_{sp}\omega_{n_{sp}} s + \omega_{n_{sp}}^2)(s^2 + 2\zeta_p \omega_{n_p} s + \omega_{n_p}^2)$$

The short period frequency $\omega_{n_{sp}}$ is usually much greater than the phugoid natural frequency ω_{n_p}. For subsonic aircraft, the short period is normally well damped, with ζ_{sp} ranging from .5 to 1. However, the phugoid is poorly damped and sometimes becomes unstable. The short period and phugoid natural frequencies and damping ratios are: **

$$\text{(III-7)} \quad \omega_{n_{sp}} \approx \sqrt{M_{\dot{\theta}} Z_w - U_0 M_w} \quad ; \quad \zeta_{sp} = -\frac{1}{2\omega_{n_{sp}}}(U_0 M_{\dot{w}} + Z_w + M_{\dot{\theta}})$$

---

* Reference 4, Equation (III-38).
** Reference 4, Table III-4.

and

$$\text{(III-8)} \qquad \omega_{n_p} \approx \frac{1}{\omega_{n_{sp}}}\sqrt{g(M_w Z_u - M_u Z_w)} \; ; \; \zeta_p \approx -\frac{X_u}{2\omega_{n_p}}$$

(d) THE EQUALIZER

The equalizer, or artificial feel system, consists of the force stability augmenter and the motion stability augmenter. The function of the artificial feel system is to effectively alter the equations of motion, (III-3), so as to improve the dynamic response of the airframe.

Consider first the force stability augmenter. The force producing system applies forces to the control stick in addition to the forces exerted by the pilot. These forces are functions of the aircraft output quantities which are fed back, either directly or indirectly, to the control stick. For the most general case, the forces can be expressed as

$$\text{(III-9)} \qquad \left[\begin{array}{l} F_f = F_u + F_{\dot{u}} + F_\theta + F_{a_z} + F_{\dot{a}_z} \\ F_f = Y_u u + Y_{\dot{u}}\dot{u} + Y_\theta \theta + Y_{a_z} a_z + Y_{\dot{a}_z}\dot{a}_z \end{array}\right.$$

where

F_f is the total force applied to the stick by the force producer

Y's are the transfer functions relating the force outputs of the force producer to the input quantities

u, $\dot{u}$, θ, a_z, and $\dot{a}_z$ are the airframe output quantities describing its dynamics.

These force feedbacks should be examined in the light of the basic purposes of force stability augmenters listed in Chapter I. These are repeated here for convenience.

1. The force producer must provide the pilot with pressure cues of the proper magnitudes and from the proper sources to allow near optimum flight path control.

2. The force producer must reduce the possibility of inadvertent destruction of the airplane.

3. Through the elevator motions produced by the force sources under hands-off flight conditions, satisfactory dynamic stability must be provided.

Consider $Y_u u$. Since u is the change in forward velocity from trim speed, the force $Y_u u$ partially satisfies the first requirement. Assume that the sensing and actuating elements which produce the force $Y_u u$ are perfect; i.e., they contain no lags. Then the transfer function Y_u can be replaced by a pure gain term, $\pm\left|K_{u_F}\right|$. The algebraic sign of K_{u_F} must be determined from physical considerations.

In Figure III-7, F_p and F_F are the forces applied by the pilot and by the force producer respectively. The positive direction of force is a pull force, corresponding to up-elevator.

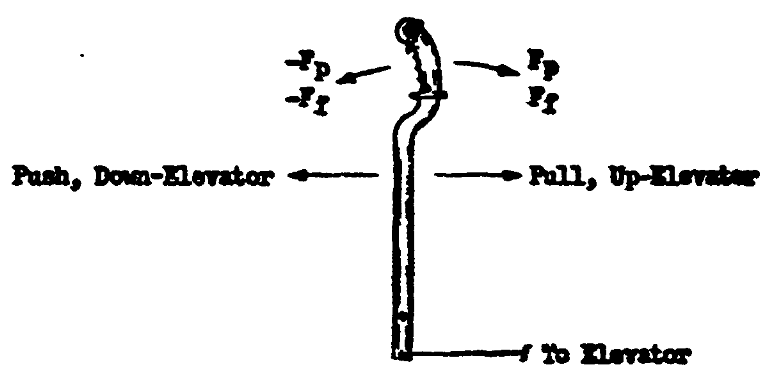


Figure III-7. Forces Acting at Stick Grip

For a statically stable airplane, up-elevator or a pull force on the control stick must be supplied to decrease speed; i.e., F_p gives $-u$. To provide the pilot with the "feel" that he is decreasing speed, $-F_F$ must act on the stick. Therefore, F_u and u should be of the same sign, or $F_u = K_{u_F} u$

(see Figure III-8).

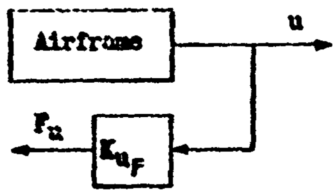


Figure III-8. Block Diagram Representing u Force Feedback

F_f effectively acts against the pilot's applied force. Then the net stick force, F_s , is

(III-10) $F_s = F_p - F_f$

This is illustrated in Figure III-9.

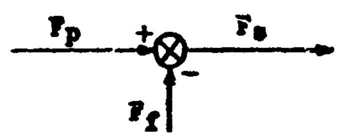


Figure III-9. Summation of Forces Acting on Stick

The net stick force acts through the control system dynamics to give an elevator deflection δ_{e_1} . Denoting the control system dynamics by the transfer function Y_{δ_e} ,

(III-11) $\delta_{e_1} = Y_{\delta_e} F_s$

If it is assumed that the control system dynamics can be represented by a simple linear spring,

(III-12) $\delta_{e_1} = \dfrac{1}{K_{\delta_e}} F_s$

as shown in Figure III-10:

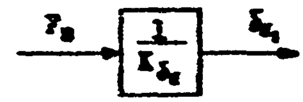


Figure III-10. Block Representing Relationship Between Elevator Motion and Stick Force

In the hands-off flight condition, F_p is zero, and δ_{E_1} (for only a u force feedback) is given by

$$\delta_{E_1} = \frac{1}{K_{\delta_E}} F_s = \frac{1}{K_{\delta_E}}(-F_f) = +\frac{1}{K_{\delta_E}}(-F_u) = \frac{-K_{u_F}}{K_{\delta_E}} u \tag{III-13}$$

The block diagram of Figure III-3 now appears as in Figure III-11:

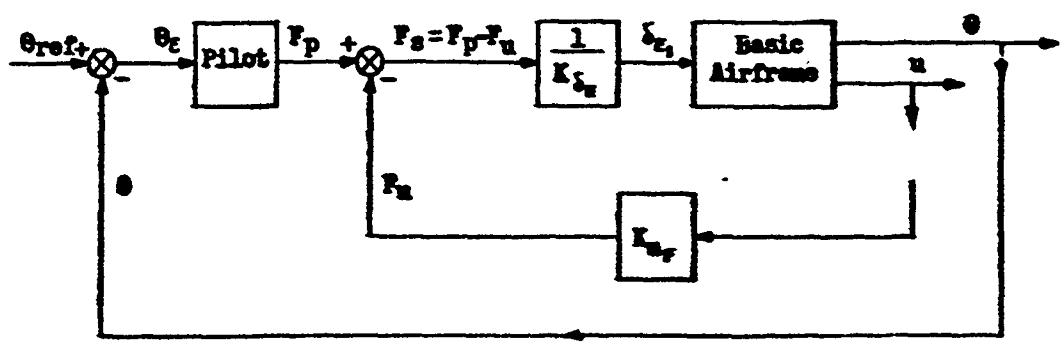


Figure III-11. Block Diagram of Figure III-3 Modified by u Force Feedback

Substituting (III-13) into (III-3) yields for the normal and pitching acceleration equations,

$$\text{(III-14)}\quad \begin{cases} a_z = Z_u u + Z_w w + Z_{\dot\theta}\dot\theta - Z_{\delta_E}\left(\frac{-K_{u_F}}{K_{\delta_E}}\right)u \\ \ddot\theta = M_u u + M_w w + M_{\dot w}\dot w + M_{\dot\theta}\dot\theta - M_{\delta_E}\left(\frac{K_{u_F}}{K_{\delta_E}}\right)u \end{cases}$$

or

$$\text{(III-15)} \quad \left[\begin{aligned} a_z &= \left(Z_u - Z_{\delta_E}\frac{K_{u_F}}{K_{\delta_E}}\right)u + Z_w w + Z_{\dot\theta}\dot\theta \\ \ddot\theta &= \left(M_u - M_{\delta_E}\frac{K_{u_F}}{K_{\delta_E}}\right)u + M_w w + M_{\dot w}\dot w + M_{\dot\theta}\dot\theta \end{aligned}\right.$$

Equation (III-15) shows that the original stability derivatives Z_u and M_u have been augmented to the new values, $Z_u - Z_{\delta_E}(K_{u_F}/K_{\delta_E})$ and $M_u - M_{\delta_E}(K_{u_F}/K_{\delta_E})$. This augmenting characteristic of the force producer leads naturally to the term "force stability augmenter."

The use of the normal acceleration a_z as a signal for the force producer satisfies the second requirement; that is, the force at the stick as felt by the pilot will build up to a large value as large load factors are built up, thus acting as a warning to the pilot.

When a pull force (up elevator) is applied to give an upward acceleration $-a_z$, the reactive force should increase as the magnitude of the acceleration increases. This reactive force is $-F_f$. Again assuming that the transfer function Y_{a_z} can be replaced by a pure gain,

$$\text{(III-16)} \quad -F_f = -F_{a_z} = -K_{a_{z_F}} a_z$$

or

$$F_{a_z} = K_{a_{z_F}} a_z$$

as shown in Figure III-12.

The forces $Y_{\dot u}\dot u$, $Y_\theta \theta$, and $Y_{\dot a_z}\dot a_z$, which are respectively proportional to the rate of change of forward velocity from trim, to the perturbation pitch angle, and to the rate of change of normal acceleration, are not particularly useful as pressure cues to the pilot. However, the quantities $\dot u$ and $\dot a_z$ are

effective augmenting systems for both hands-off and hands-on flying, but for augmenting purposes alone, these quantities are better utilized in the motion stability augmenter.

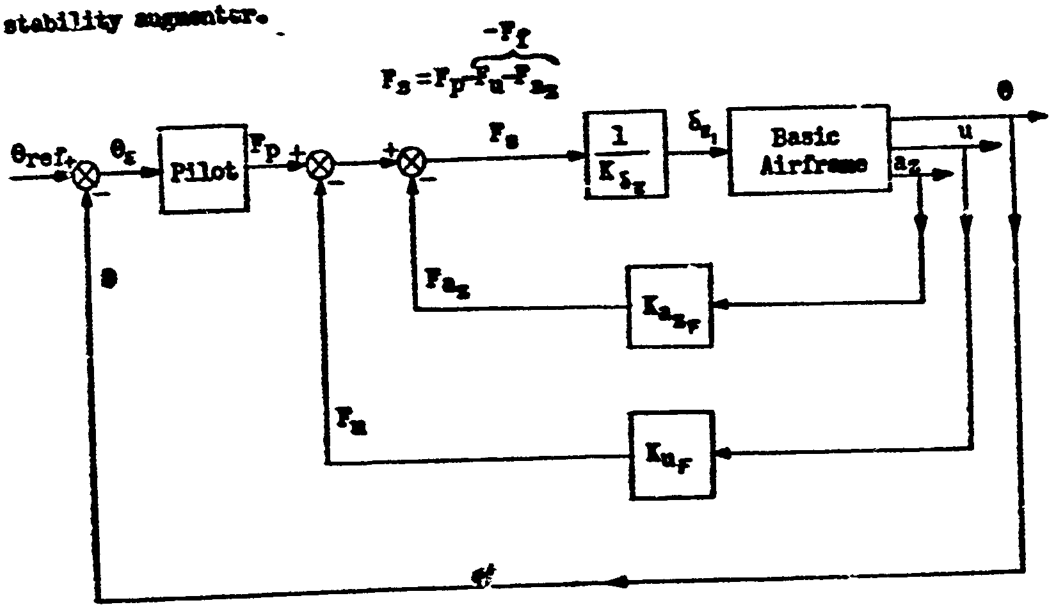


Figure III-12. Block Diagram of Figure III-11 Modified by a_z Force Feeback

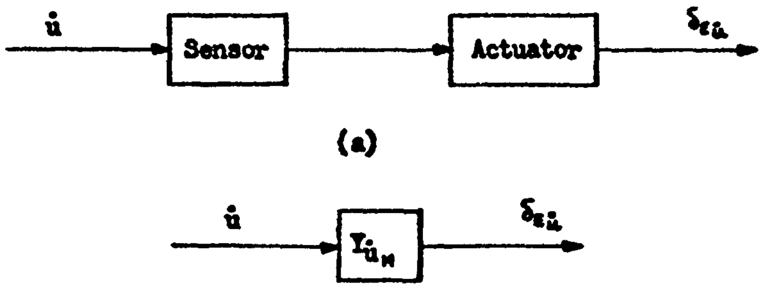
The motion stability augmenter automatically deflects the elevator so as to stabilize the characteristic modes of the airframe. These elevator deflections are superimposed on the elevator deflections caused by the pilot or the force stability augmenter.

Present experience has indicated that four of the most useful airframe quantities employed in automatic motion stability augmentation are u, $\dot{u}$, a_z, and $\dot{a}_z$. Since the phugoid mode of the airframe is largely a matter of airspeed changes, it seems feasible to use u and $\dot{u}$ to improve the characteristics of this mode. The damping of the phugoid is best done by $\dot{u}$ feedback while u feedback effectively eliminates the "tuck-under" tendencies associated with flight through the transonic region.

Consider the $\dot{u}$ feedback. The elevator deflection for this feedback is

$$\text{(III-17)} \qquad \delta_{E_{\dot u}} = Y_{\dot u_M}\, \dot u$$

where $Y_{\dot u_M}$ is the transfer function relating elevator deflection to $\dot u$. This is illustrated in Figure III-13.



(b)

Figure III-13. Block Diagram of ů Feedback to Elevator

Assuming no lags in $Y_{\dot u_M}$,

$$\text{(III-18)} \qquad \delta_{E_{\dot u}} = K_{\dot u_M}\, \dot u$$

Substituting (III-18) into (III-3),

$$\text{(III-19)} \qquad \begin{cases} a_z = Z_u u + Z_w w + Z_{\dot\theta}\dot\theta + Z_{\delta_E} K_{\dot u_M} \dot u \\ \ddot\theta = M_u u + M_w w + M_{\dot w}\dot w + M_{\dot\theta}\dot\theta + M_{\delta_E} K_{\dot u_M}\dot u \end{cases}$$

Thus $\dot u$ feedback creates two new stability derivatives, $Z_{\delta_E} K_{\dot u_M}$ and $M_{\delta_E} K_{\dot u_M}$. These stability derivatives change the B, C, and D coefficients in (III-4) to

$$\text{(III-20)*} \qquad \begin{cases} B' = B - X_w Z_{\delta_E} K_{\dot u_M} \\ C' = C + Z_{\delta_E} K_{\dot u_M}(X_w M_{\dot\theta} + g M_{\dot w}) + M_{\delta_E} K_{\dot u_M}(g - X_w U_0) \\ D' = D + g(M_w Z_{\delta_E} K_{\dot u_M} - Z_w M_{\delta_E} K_{\dot u_M}) \end{cases}$$

* See Appendix for derivations.

where B', C', and D' are the new coefficients. B, C, and D are usually positive numbers. Denoting B', C', and D' as

$$\text{(III-21)} \quad \begin{cases} B' = B + \Delta B \\ C' = C + \Delta C \\ D' = D + \Delta D \end{cases}$$

the phugoid quadratic can be rewritten as

$$\text{(III-22)} \quad \left[s^2 + \frac{(D+\Delta D)(C+\Delta C)-(B+\Delta B)E}{(C+\Delta C)^2}\, s + \frac{E}{C+\Delta C}\right]$$

or

$$\left[s^2 + \frac{(DC-BE)+C\Delta D + D\Delta C + \Delta D\Delta C - E\Delta B}{(C+\Delta C)^2}\, s + \frac{E}{C+\Delta C}\right] =$$

$$\left(s^2 + 2\zeta_p' \omega_{n_p}' s + \omega_{n_p}'^2\right)$$

Equation (III-22) shows that

$$\text{(III-23)} \quad \begin{cases} 2\zeta_p' \omega_{n_p}' > 2\zeta_p \omega_{n_p} \\ \omega_{n_p}'^2 < \omega_{n_p}^2 \end{cases}$$

if ΔB, ΔC, and ΔD are positive. Then from the two inequalities in (III-23), it can be seen that $\zeta_p' > \zeta_p$, thus showing that $\dot{u}$ feedback increases the phugoid damping.

Consider now the u feedback. As in the $\dot{u}$ case,

$$\text{(III-24)} \quad \delta_{E_u} = Y_{u_M} u$$

or

(III-25) $\quad \delta_{E_u} = K_{u_M} u$

for a perfect sensor and actuator as shown in Figure III-14.

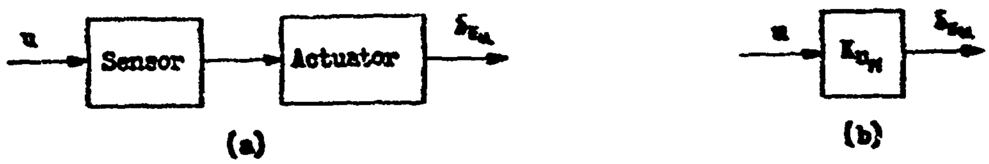


Figure III-14. Block Diagram of u Feedback to Elevator

Substituting (III-25) into (III-3),

$$\text{(III-26)} \quad \begin{cases} a_z = (Z_u + Z_{\delta_E} K_{u_M})u + Z_w w + Z_{\dot\theta}\dot\theta \\ \ddot\theta = (M_u + M_{\delta_E} K_{u_M})u + M_w w + M_{\dot w}\dot w + M_{\dot\theta}\dot\theta \end{cases}$$

Equations (III 26) show the augmenting value of u feedback. This augmentation is most useful for eliminating the tuck-under. Tuck-under occurs when the E coefficient of (III-4) becomes negative, in which case ω_{n_p} becomes an imaginary number. E is given by

(III-27) $\quad E = g(Z_u M_w - M_u Z_w)$

With the augmented values of Z_u and M_u,

$$\text{(III-28)} \quad \begin{aligned} E' &= g[(Z_u + Z_{\delta_E} K_{u_M})M_w - (M_u + M_{\delta_E} K_{u_M})Z_w] \\ &= g[Z_u M_w - Z_w M_u] + g[Z_{\delta_E} K_{u_M} M_w - M_{\delta_E} K_{u_M} Z_w] \\ &= E + \Delta E \end{aligned}$$

By making the proper choice of K_{u_M}, E' can be made positive, thus making ω_{n_p} a real number and eliminating the tuck-under tendency.

Since the elevator deflections caused by the motion stability augmenter are superimposed on the elevator deflections caused by the pilot or the force stability augmenter,

(III-29) $\delta_e = \delta_{e_1} + \delta_{e_M}$

where

δ_e is the total elevator deflection

δ_{e_1} is the elevator deflection caused by the pilot and the force stability augmenter

δ_{e_M} is the elevator deflection caused by the motion stability augmenter

Figure III-12 should now be further modified to include the elevator contribution from the u and $\dot{u}$ motion stability augmentation feedbacks.

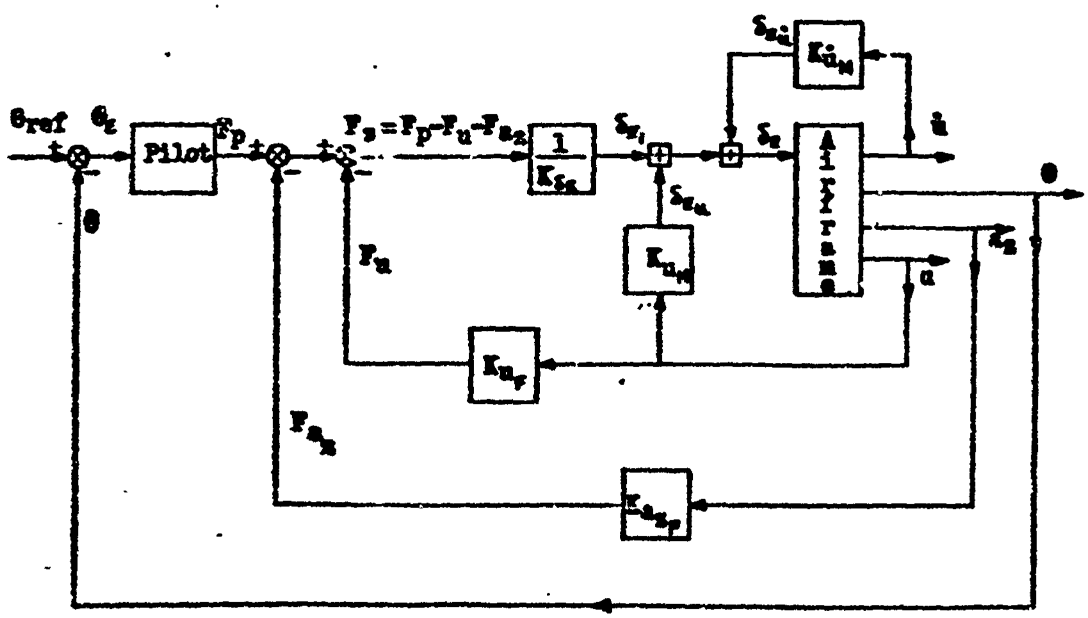


Figure III-15. Block Diagram of Figure III-12 Modified by u and u̇ Feedbacks to Elevator

Very little change occurs in airspeed during the short period mode, and consequently, neither u nor $\dot{u}$ feedback will have much effect on the short period. However, large variations in a_z and $\dot{a}_z$ are exhibited in the short period, and these quantities give excellent control of this mode.

The elevator deflections caused by a_z and $\dot{a}_z$ feedback are

(III-30) $$\delta_{E_{a_z}} + \delta_{E_{\dot{a}_z}} = Y_{a_{z_M}} a_z + Y_{\dot{a}_{z_M}} \dot{a}_z$$

Again assuming perfect sensors and actuators,

(III-31) $$\delta_{E_{a_z}} + \delta_{E_{\dot{a}_z}} = K_{a_{z_M}} a_z + K_{\dot{a}_{z_M}} \dot{a}_z$$

This augmentation of the elevator motions by a_z and $\dot{a}_z$ causes the following changes in the coefficients of (III-4):

(III-32) $$\Delta' = A_1 s^5 + A' s^4 + B' s^3 + C' s^2 + D' s + E$$

where

$$A_1 = A_1$$

$$A' = A + \Delta A_{a_z} + \Delta A_{\dot{a}_z}$$

$$B' = B + \Delta B_{a_z} + \Delta B_{\dot{a}_z}$$

$$C' = C + \Delta C_{a_z} + \Delta C_{\dot{a}_z}$$

$$D' = D + \Delta D_{a_z} + \Delta D_{\dot{a}_z}$$

Note that the order of the characteristic equation is increased and that the E coefficient is unchanged. The augmented coefficients from the a_z and

$\ddot{a}_z$ feedbacks are

$$
\text{(III-33)*}\quad
\begin{cases}
\Delta A_{a_z} = -Z_{\delta_E} K_{a_{z_M}} \\
\Delta B_{a_z} \approx Z_{\delta_E} K_{a_{z_M}} M_{\dot\theta} - M_{\delta_E} K_{a_{z_M}} Z_{\dot\theta} \\
\Delta C_{a_z} \approx Z_{\delta_E} K_{a_{z_M}} (U_0 M_w - M_{\dot\theta} X_u) - M_{\delta_E} K_{a_{z_M}} U_0 Z_w \\
\Delta D_{a_z} \approx Z_{\delta_E} K_{a_{z_M}} \left[U_0 (M_u X_w - X_u M_w) - M_u g\right] + M_{\delta_E} K_{a_{z_M}} \left[U_0 (X_u Z_w - X_w Z_u) + Z_u g\right]
\end{cases}
$$

and

$$
\text{(III-34)}\quad
\begin{cases}
A_1 \approx -Z_{\delta_E} K_{\dot a_{z_M}} \\
\Delta A_{\dot a_z} \approx Z_{\delta_E} K_{\dot a_{z_M}} M_{\dot\theta} - M_{\delta_E} K_{\dot a_{z_M}} Z_{\dot\theta} \\
\Delta B_{\dot a_z} \approx Z_{\delta_E} K_{\dot a_{z_M}} (U_0 M_w - M_{\dot\theta} X_u) - M_{\delta_E} K_{\dot a_{z_M}} U_0 Z_w \\
\Delta C_{\dot a_z} \approx Z_{\delta_E} K_{\dot a_{z_M}} \left[U_0 (X_w M_u - X_u M_w) - M_u g\right] + M_{\delta_E} K_{\dot a_{z_M}} \left[U_0 (X_u Z_w - X_w Z_u) + Z_u g\right]
\end{cases}
$$

Equation (III-7) shows that the short period frequency and damping ratio are

$$\omega_{n_{sp}} \approx \sqrt{C}$$

$$\zeta_{sp} \approx \frac{B}{2\sqrt{C}}$$

By the proper choice of $K_{a_{z_M}}$ and $K_{\dot a_{z_M}}$ in (III-33) and (III-34), it is possible to increase $\omega_{n_{sp}}$ while holding ζ_{sp} constant or increasing it.

The complete generalized block diagram of the pilot-airframe system is shown in Figure III-16.

* See Appendix for derivation.

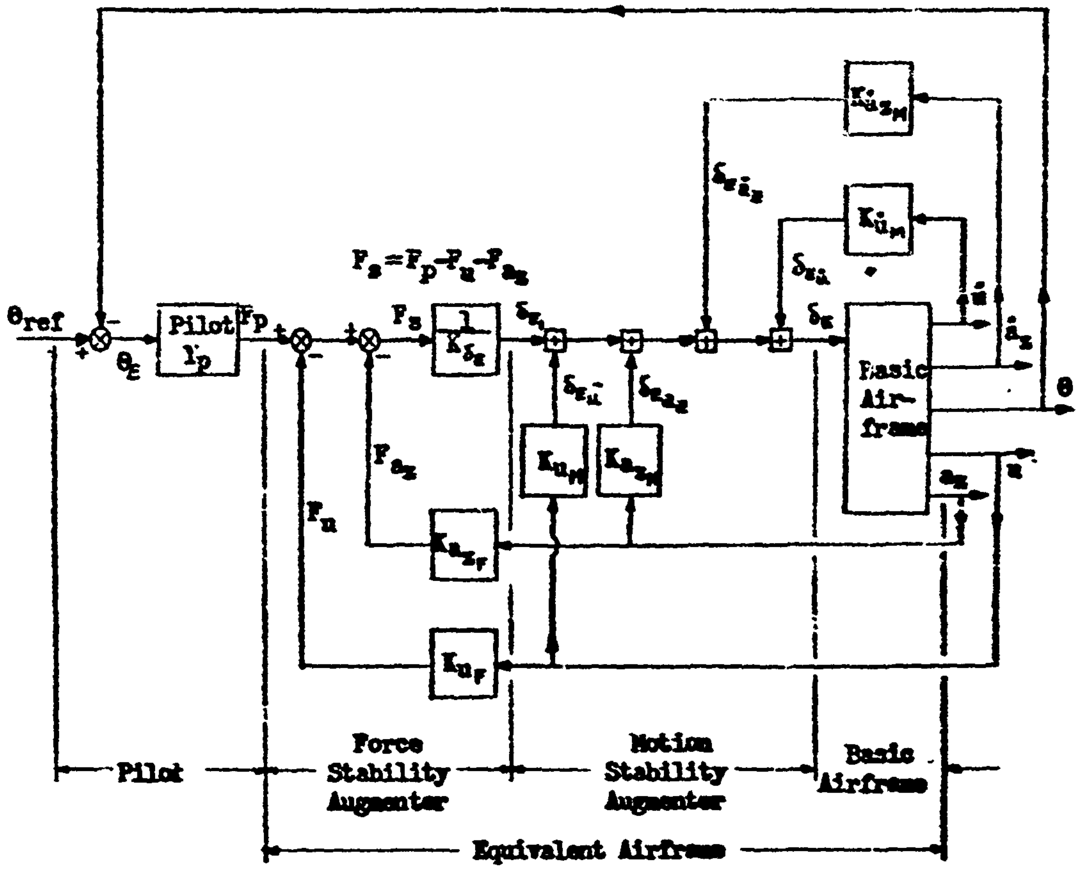


Figure III-16. Generalized Block Diagram of Pilot-Equivalent Airframe System

All pertinent equations are summarized here for convenience.

<u>The controller</u> (the pilot):

(III-35) $$F_p = Y_p \theta_\varepsilon = K e^{-\tau s}\left(\frac{T_1 s + 1}{T_2 s + 1}\right)\theta_\varepsilon$$

<u>The equalizer:</u>

The force stability augmenting system:

(III-36) $$\begin{cases} F_s = F_p - F_f \\ F_f = F_u + F_{a_z} \\ F_u = + K_{u_F} u \end{cases}$$

(III-36)
(Cont.)

$$\begin{cases} F_{a_z} = + K_{a_{z_F}} a_z \\ \delta_{e_1} = \dfrac{1}{K_{\delta_z}} F_z \end{cases}$$

The motion stability augmenting system:

$$\text{(III-37)} \quad \begin{cases} \delta_e = \delta_{e_1} + \delta_{e_M} \\ \delta_{e_M} = \delta_{e_u} + \delta_{e_{\dot{u}}} + \delta_{e_{a_z}} + \delta_{e_{\dot{a}_z}} \\ \delta_{e_u} = K_{u_M} u \\ \delta_{e_{\dot{u}}} = K_{\dot{u}_M} \dot{u} \\ \delta_{e_{a_z}} = K_{a_{z_M}} a_z \\ \delta_{e_{\dot{a}_z}} = K_{\dot{a}_{z_M}} \dot{a}_z \end{cases}$$

The controlled element (the airframe):

$$\text{(III-38)} \quad \begin{cases} \dot{u} = X_u u + X_w w - g\theta \\ \dot{w} - U_0 \dot{\theta} = a_z = Z_u u + Z_w w + Z_{\dot{\theta}} \dot{\theta} + Z_{\delta_e} \delta_e \\ \ddot{\theta} = M_u u + M_w w + M_{\dot{w}} \dot{w} + M_{\dot{\theta}} \dot{\theta} + M_{\delta_e} \delta_e \end{cases}$$

(e) THE EQUIVALENT AIRFRAME AND EQUIVALENT STABILITY DERIVATIVES

In Figure III-16, the combination of the equalizing system and the controlled element are referred to as the equivalent airframe. This is the equivalent airframe indicated in Figure III-3. The equations of the equivalent airframe can be derived from Figure III-16 or from Equations (III-35), (III-36), (III-37), and (III-38).

From (III-36) and (III-37), the total elevator deflection δ_E is

$$\text{(III-39)} \qquad \delta_E = \frac{1}{K_{\delta_E}}\left(Y_p \theta_E - K_{u_E} u - K_{a_{z_E}} a_z\right) + K_{u_M} u + K_{\dot{u}_M} \dot{u} + K_{a_{z_M}} a_z + K_{\dot{a}_{z_M}} \dot{a}_z$$

or denoting $Y_p \theta_E / K_{\delta_E}$ as the elevator deflection δ_{E_p} due to the pilot's efforts alone,

$$\text{(III-40)} \qquad \delta_E = \delta_{E_p} + \left(\frac{-K_{u_E}}{K_{\delta_E}} + K_{u_M}\right)u + K_{\dot{u}_M}\dot{u} + \left(-\frac{K_{a_{z_E}}}{K_{\delta_E}} + K_{a_{z_M}}\right)a_z + K_{\dot{a}_{z_M}}\dot{a}_z$$

$$\text{(III-40a)} \qquad \delta_{E_p} = \frac{1}{K_{\delta_E}} F_p = \frac{1}{K_{\delta_E}} Y_p \theta_E = \frac{K}{K_{\delta_E}} e^{-\tau s}\left(\frac{T_1 s + 1}{T_2 s + 1}\right)\theta_E$$

Substituting (III-40) into (III-38),

$$\text{(III-41)} \qquad \left[\begin{aligned}
&\dot{u} = X_u u + X_w w - g\theta \\
&\dot{w} - U_0\dot{\theta} = a_z = \left[Z_u + Z_{\delta_E}\left(-\frac{K_{u_E}}{K_{\delta_E}} + K_{u_M}\right)\right]u + Z_{\delta_E} K_{\dot{u}_M}\dot{u} + Z_w w + Z_{\dot{\theta}}\dot{\theta} \\
&+ Z_{\delta_E}\left(-\frac{K_{a_{z_E}}}{K_{\delta_E}} + K_{a_{z_M}}\right)a_z + Z_{\delta_E} K_{\dot{a}_{z_M}}\dot{a}_z + Z_{\delta_E}\delta_{E_p} \\
&\ddot{\theta} = \left[M_u + M_{\delta_E}\left(-\frac{K_{u_E}}{K_{\delta_E}} + K_{u_M}\right)\right]u + M_{\delta_E} K_{\dot{u}_M}\dot{u} + M_w w + M_{\dot{w}}\dot{w} + M_{\dot{\theta}}\dot{\theta} \\
&+ M_{\delta_E}\left(-\frac{K_{a_{z_E}}}{K_{\delta_E}} + K_{a_{z_M}}\right)a_z + M_{\delta_E} K_{\dot{a}_{z_M}}\dot{a}_z + M_{\delta_E}\delta_{E_p}
\end{aligned}\right.$$

The equations (III-41) are the equations of motion of the equivalent airframe. It is with respect to this equivalent airframe that optimum design is attempted. That is, the responses u, w, θ, etc., must be optimized bearing in mind that the input to the equivalent airframe is δ_{E_p}, or more exactly F_p, the pilot's force output.

The augmented, or equivalent stability derivatives are summarized in Table III-1. Note particularly that the relationship between the augmented M and Z derivatives are given by the constant $K_1 = M_{\delta_E}/Z_{\delta_E}$; that is,

$$
\text{(III-42)} \qquad \left[\begin{aligned} M_u' &= K_1 Z_u' \\ M_{\dot u} &= K_1 Z_{\dot u} \\ M_{a_z} &= K_1 Z_{a_z} \\ M_{\dot a_z} &= K_1 Z_{\dot a_z} \end{aligned} \right.
$$

Equivalent Stability Derivatives	=	Basic Airframe Stability Derivatives	+	Augmented Stability Derivatives From Force Stability Augmenter	+	Augmented Stability Derivatives From Motion Stability Augmenter	=	Alternate Expression
Z_{u_T}	=	Z_u	−	$Z_{\delta_E} \frac{K_{u_F}}{K_{\delta_E}}$	+	$Z_{\delta_E} K_{u_M}$	=	$Z_u + Z_u'$
$Z_{\dot u}$	=	0	+	0	+	$Z_{\delta_E} K_{\dot u_M}$	=	$Z_{\dot u}$
Z_{a_z}	=	0	−	$Z_{\delta_E} \frac{K_{a_{zF}}}{K_{\delta_E}}$	+	$Z_{\delta_E} K_{a_{zM}}$	=	Z_{a_z}
$Z_{\dot a_z}$	=	0	+	0	+	$Z_{\delta_E} K_{\dot a_{zM}}$	=	$Z_{\dot a_z}$
M_{u_T}	=	M_u	−	$M_{\delta_E} \frac{K_{u_F}}{K_{\delta_E}}$	+	$M_{\delta_E} K_{u_M}$	=	$M_u + M_u'$
$M_{\dot u}$	=	0	+	0	+	$M_{\delta_E} K_{\dot u_M}$	=	$M_{\dot u}$
M_{a_z}	=	0	−	$M_{\delta_E} \frac{K_{a_{zF}}}{K_{\delta_E}}$	+	$M_{\delta_E} K_{a_{zM}}$	=	M_{a_z}
$M_{\dot a_z}$	=	0	+	0	+	$M_{\delta_E} K_{\dot a_{zM}}$	=	$M_{\dot a_z}$

Table III-1. Equivalent Stability Derivatives

Then (III-41) is simplified to

$$
\text{(III-43)} \quad \begin{cases} \dot{u} = X_u u + X_w w - g\Theta \\ \dot{w} - U_0\dot{\theta} = a_z = Z_u u + Z_w w + Z_{\dot{\theta}}\dot{\theta} + Z_{\delta_z}\delta_{\epsilon_p} + (Z'_u u + Z_{\dot{u}}\dot{u} + Z_{a_z}a_z + Z_{\dot{a}_z}\dot{a}_z) \\ \ddot{\theta} = M_u u + M_w w + M_{\dot{w}}\dot{w} + M_{\dot{\theta}}\dot{\theta} + M_{\delta_z}\delta_{\epsilon_p} + K_1(Z'_u u + Z_{\dot{u}}\dot{u} + Z_{a_z}a_z + Z_{\dot{a}_z}\dot{a}_z) \end{cases}
$$

Examination of (III-43) shows that the only variables over which the designer has any great degree of control are Z'_u , $Z_{\dot{u}}$, Z_{a_z} , $Z_{\dot{a}_z}$, and K_{δ_z} for the particular example chosen. These five variables define the equalizing artificial feel system. Therefore, the problem of control system design reduces to that of adjusting these five parameters so that optimum over-all system response is obtained.

The number of variable parameters and the complexity of the many defining equations, such as (III-4), (III-7), (III-8), (III-20), (III-28), (III-33), and (III-34), seem to indicate that the design problem is not a simple one. An attempt at solving the problem through paper analysis will undoubtedly prove this to be true. Therefore, the problem is best solved by making use of analog computers. Using this method, it is only necessary to vary the values of the components corresponding to Z'_u , $Z_{\dot{u}}$, Z_{a_z} , $Z_{\dot{a}_z}$, and K_{δ_z} until the best system response is observed on the computer recorders.

SECTION 3 – THE ANALOG COMPUTER

The system equations to be set up on the computer are (III-2), (III-40a), and (III-43). To reduce the number of variable potentiometers in the computer investigation, (III-43) can be further simplified as shown in (III-44).

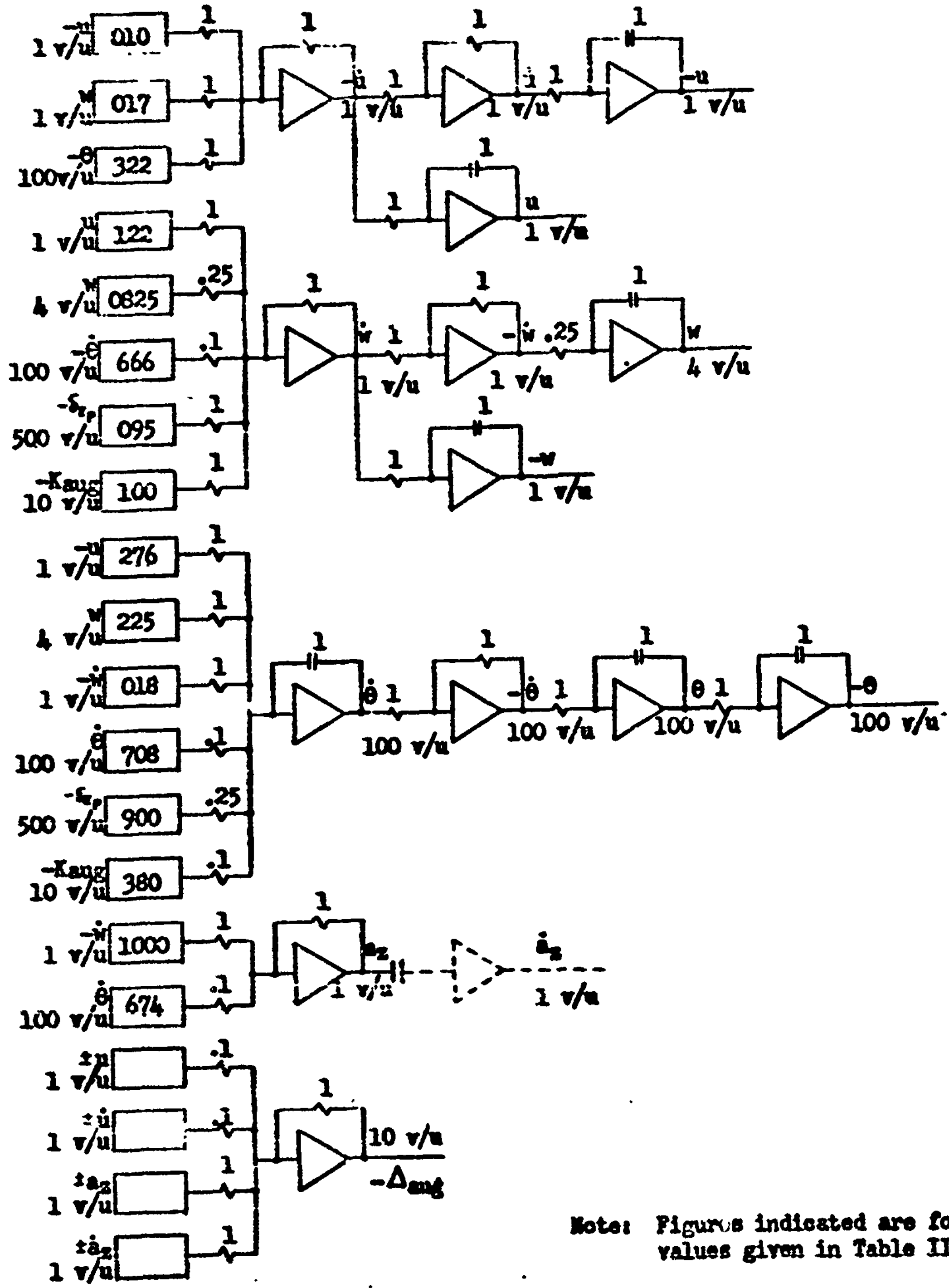
$$
\text{(III-44)} \quad \left[\begin{array}{l}
\dot{u} = X_u u + X_w w - g\theta \\[2ex]
\dot{w} = Z_u u + Z_w w + (U_0 + Z_{\dot\theta})\dot{\theta} + Z_{\delta_E} \delta_{E_p} + K_{AUG} \\[2ex]
\ddot{\theta} = M_u u + M_w w + M_{\dot w}\dot{w} + M_{\dot\theta}\dot{\theta} + M_{\delta_E}\delta_{E_p} + K_1 K_{AUG} \\[2ex]
a_z = \dot{w} - U\dot{\theta} \\[2ex]
K_{AUG} = Z'_u u + Z_{\dot u}\dot{u} + Z_{a_z} a_z + Z_{\dot a_z}\dot{a}_z
\end{array}\right.
$$

The computer circuitry for the above equations is shown in Figure III-17. Notice that since the potentiometer settings for the basic airframe stability derivatives are predetermined by the particular airframe configuration chosen, the number of variable potentiometer settings is only four.

Furthermore, it is known that two of these potentiometer settings, Z'_u and $Z_{\dot u}$, ordinarily affect only the phugoid response; occasionally they may have a slight influence on the short period mode when the frequencies of the two modes are not greatly separated. Also, Z_{a_z} and $Z_{\dot a_z}$ are chiefly short period augmenting variables.

The analysis made above shows that the complexity of the problem as originally presented can be greatly reduced by making use of equivalent stability derivatives and analog computer simulation. The analog computer program for the solution of the problem is straightforward and is presented briefly below.

Figure III-18 is the time history of a typical airframe disturbed by a gust of wind. The airframe parameters are given in Table III-2 for this particular condition.



Note: Figures indicated are for values given in Table III-2

Figure III-17. Simulation of Aircraft Equations of Motion

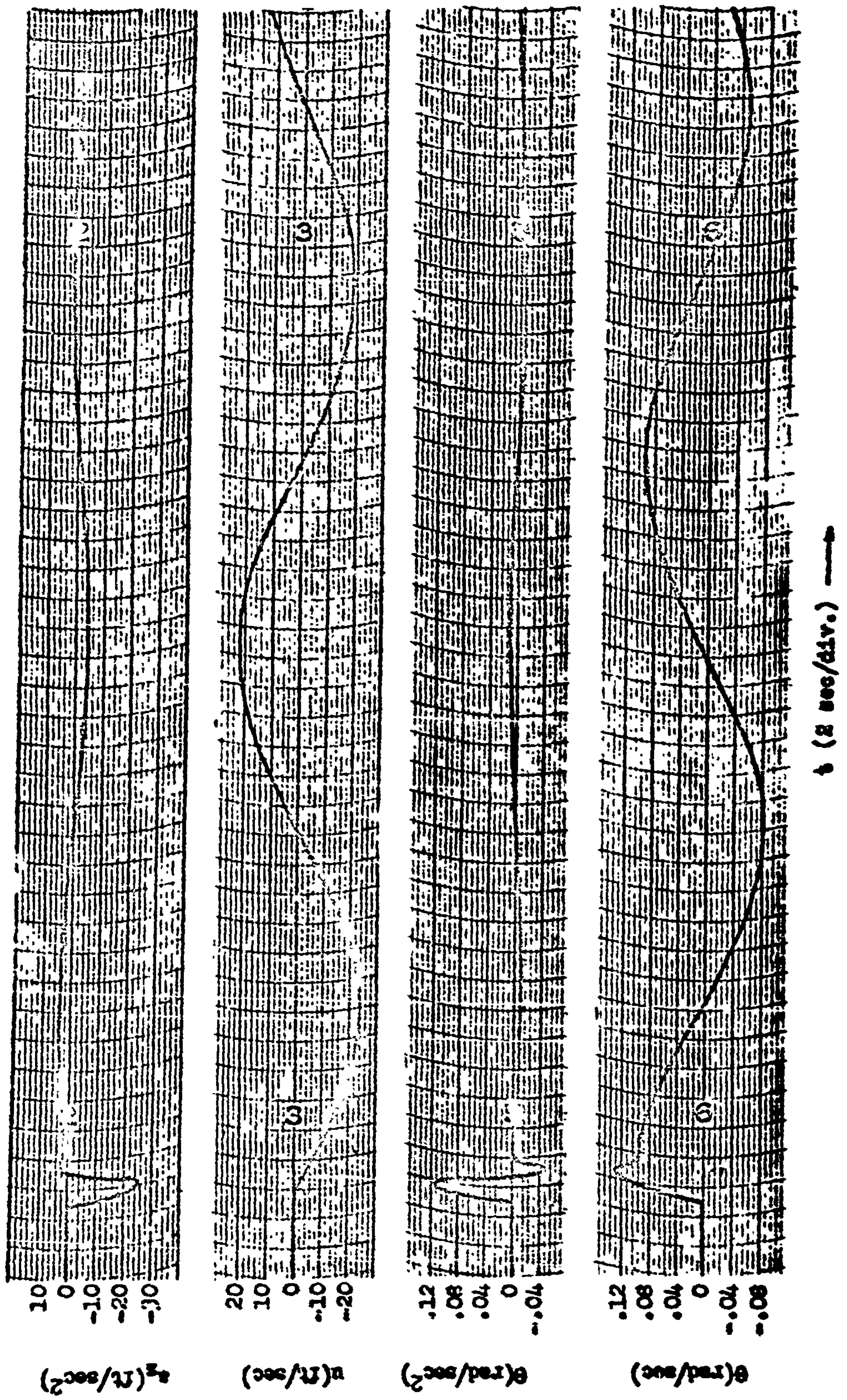


Figure III-18. Response of Unaugmented Airframe to [illegible] Gust Input for Conditions of Table III-2.

$U_0 = 674$	$X_u = -.0099$
$M_u = .00276$	$X_w = .0169$
$M_w = -.009$	$Z_u = -.122$
$M_{\dot{w}} = .00018$	$Z_w = -1.32$
$M_{\dot{\theta}} = -2.83$	$Z_{\dot{\theta}} = -8$
$M_{\delta_E} = 18.02$	$Z_{\delta_E} = 47.5$

Table III-2. Airframe Parameters Used in Figures III-18 through III-27

The time histories show that for this configuration the airframe short period mode is well damped whereas the phugoid mode is poorly damped. The phugoid period is approximately 50 seconds. It is known that a pilot can usually control oscillations of this type with little difficulty.

Figure III-19 is the time history of a pilot-basic airframe combination.* The shape of the θ response indicates that some sort of augmentation is

---

* Ordinarily, for a step θ_{ref} command input, the "compute switch" of the analog computer is turned on, and the step function is introduced. However, on the particular equipment used, this method gives rise to switching transients. To avoid these transients, the step function is applied before the "compute switch" is turned on. Because of this, the initial values of the traces for θ_E , F_p , and δ_{E_p} are offset from the usual zero reference line; i.e., $\theta_E(0) \neq 0$, $F_p(0) \neq 0$, and $\delta_{E_p}(0) \neq 0$.

This means that initially, the pilot is applying a control force creating an error in pitch attitude. At some instant, $t = 0$, the pilot receives a command to reduce this error to zero. In the steady state, θ_E never reaches zero, i.e., $\theta_{E_{ss}} \neq 0$, because the system is not a zero position error system and because of the threshold and deadband effects incorporated in the circuit.

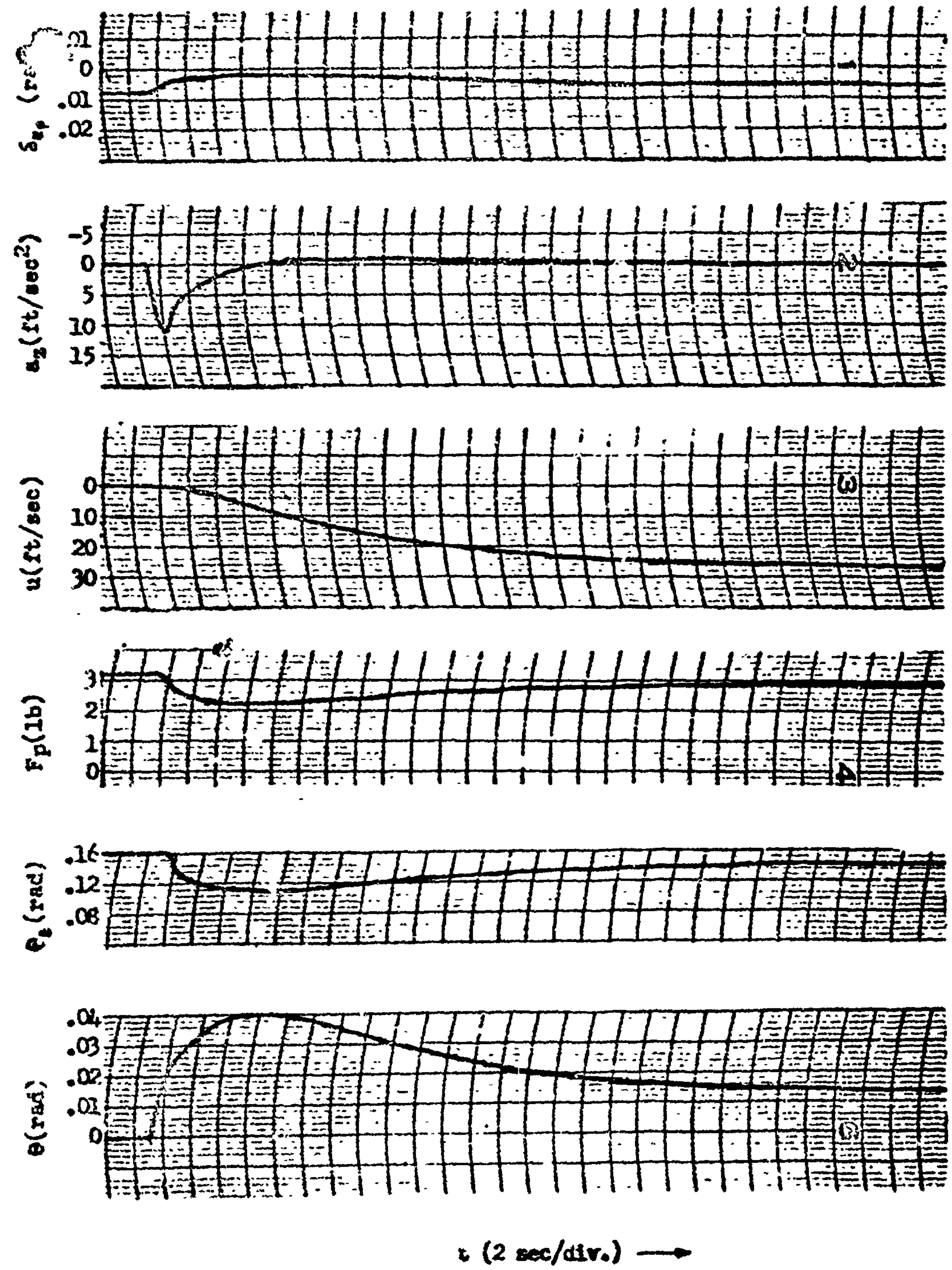


Figure III-19. Response of Pilot-Augmented Airframe System to Step θ_{ref} (.16 rad) Command for Conditions of Table III-2 ($K = 24$ lb/rad, $\tau = 0.3$ sec, $T_1 = 0$, $T_2 = 0.2$ sec).

required to eliminate the large drop in θ. The effects of augmentation will be shown in another example, but for the moment, consider the influence of the pilot on the system dynamics.

The pilot transfer function was given as

$$Y_p = Ke^{-\tau s}\left(\frac{T_1 s + 1}{T_2 s + 1}\right).$$

The computer circuitry for this transfer function is indicated in Figure III-20. Provisions are made for variation of K and T_1 since these constants are most apt to vary from one pilot to another. The pilot indifference threshold and a control system deadband are also included. The time histories in Figure III-19 are for $K = 24$ lb/rad. and $T_1 = 0$. Setting $T_1 = 0$ signifies that the pilot is flying strictly by position, i.e., by the magnitude of θ_E, and is not using the rate of change of θ_E.

Figure III-21 shows the effect of using some rate judgment $(T_1 = 1)$. The most significant effect is a slight decrease in the short period damping ratio. This is even more evident in Figure III-22 where the rate judgment time constant has been made equal to 2. The plot of normal acceleration a_z shows that the short period oscillation is not completely damped until 5 or 6 cycles have been completed. The decrease in damping indicates that the pilot is tending to overcontrol the system at short period frequencies.

Figure III-23 shows that increasing T_1 to 3 seconds has a drastic effect on the short period response. The time histories indicate that the pilot, in attempting to control the motions, overcontrols the short period mode and builds up the short period oscillations to a point where the system becomes unstable. Of course, no pilot would continue his efforts as long as is shown in Figure III-23 (approximately 45 seconds) but would release the stick and begin anew.

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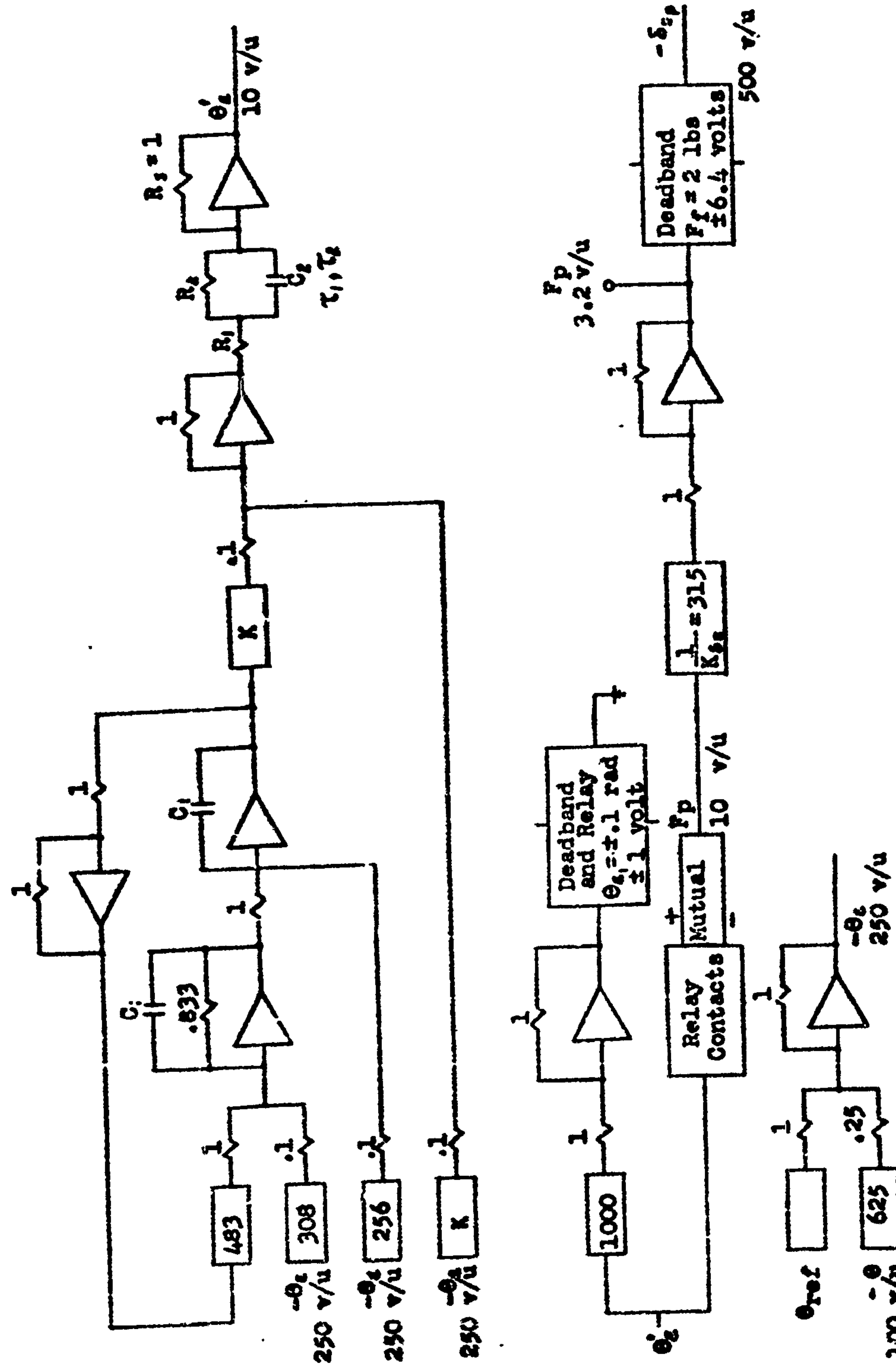

Figure III-20. Pilot Simulation

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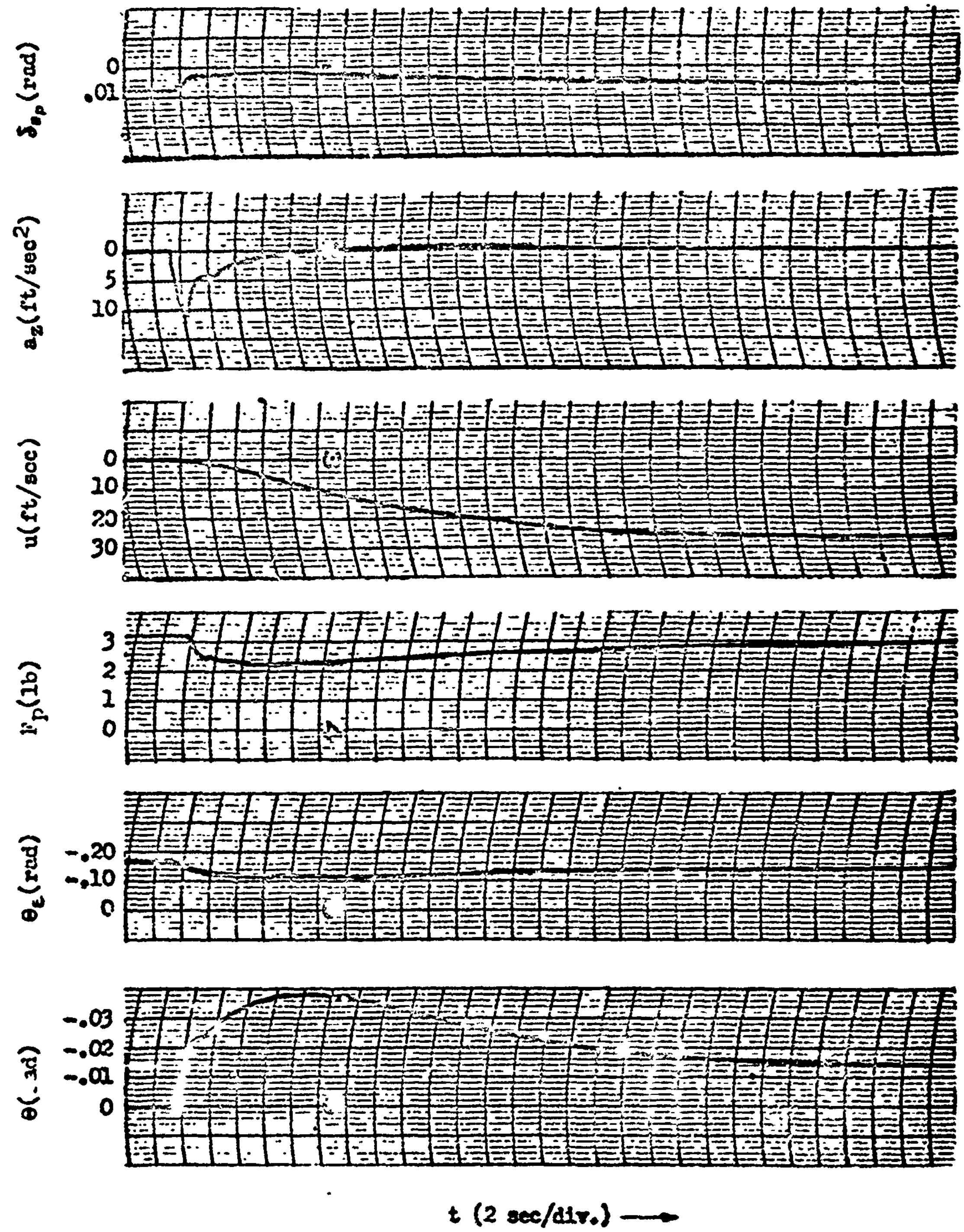


Figure III-21. Response of Pilot-Unaugmented Airframe System to Step θ_{ref} Command for Same Conditions as Figure III-19 except T_I= 1 sec.

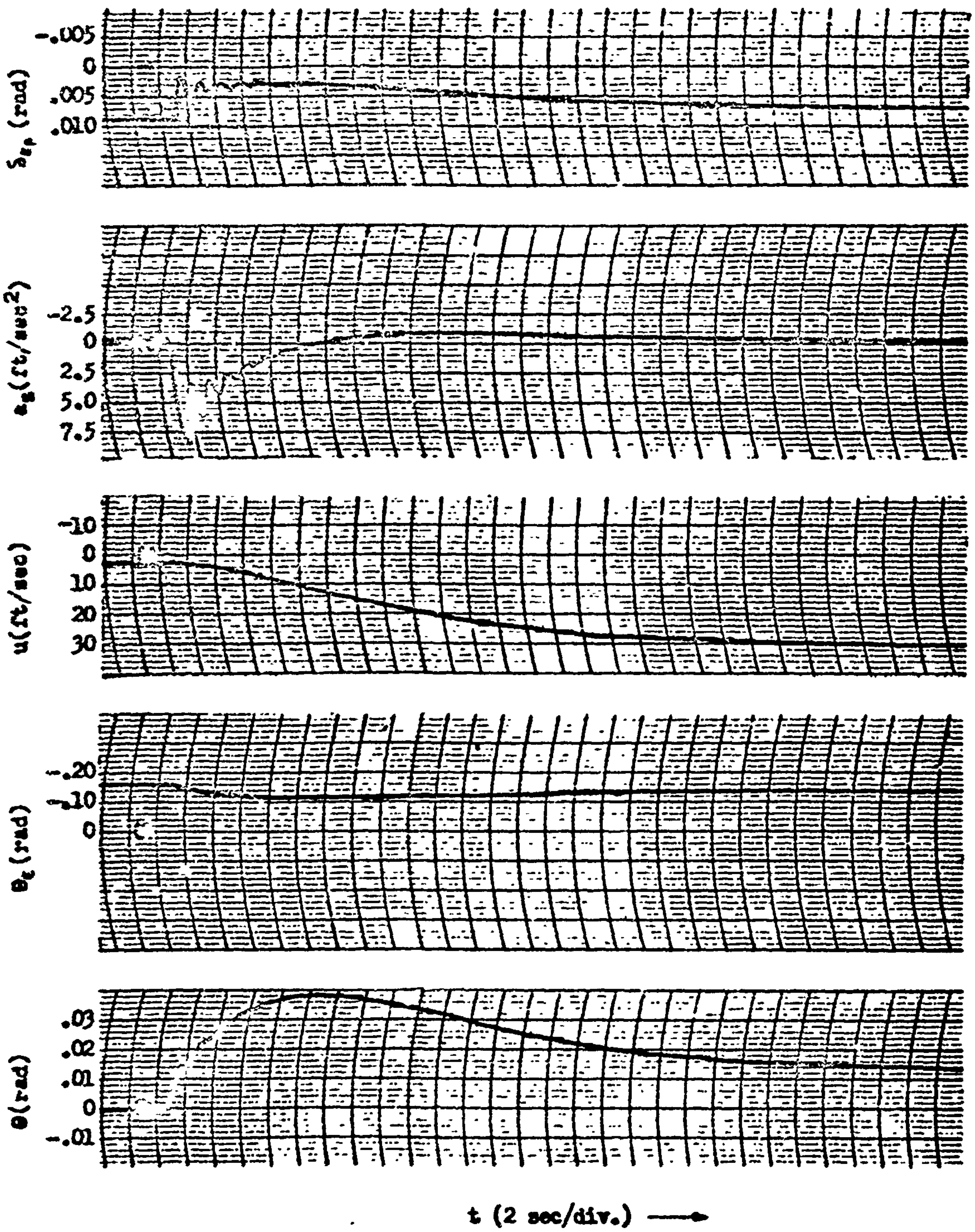


Figure III-22. Response of Pilot-Unaugmented Airframe System to Step θ_{ref} Command for Same Conditions as Figure III-19 except T_I = 2 sec.

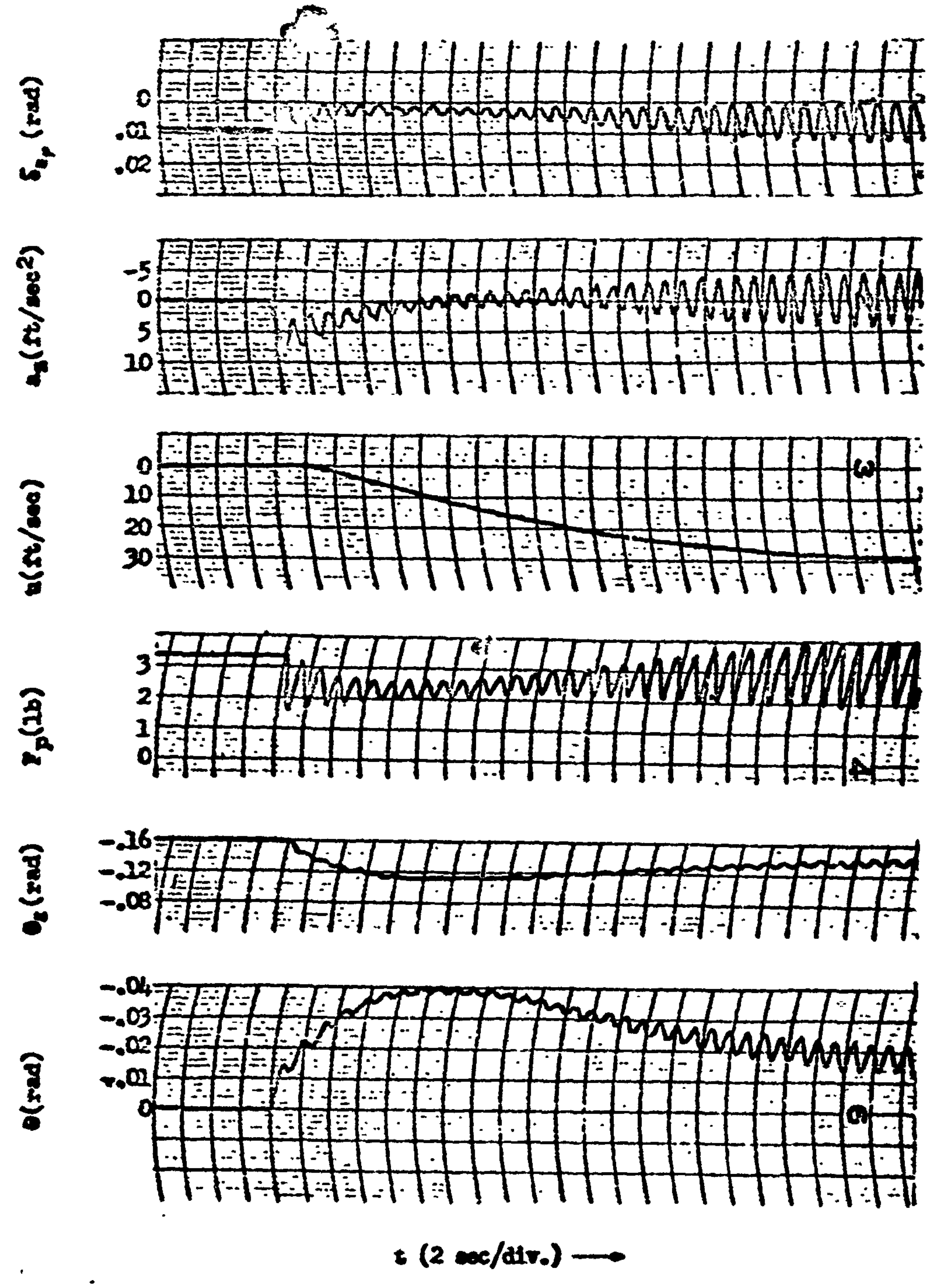


Figure III-23. Response of Pilot-Unaugmented Airframe System to Step θ_{ref} Command for Same Conditions as Figure III-19 except $T_1 = 3$ sec.

Figures III-24 and III-25 again show the effect of increasing the rate judgment time constant, in this instance with the pilot gain set at 40 lb/rad. For this higher gain, the system becomes unstable at $T_1 = 2$.

Figures III-26 and III-27 are for the pilot gain $K = 48$ lb/rad., with $T_1 = 0$ and 1 second respectively. For the same values of time constant T_1, the high frequency damping is less than for the case in which $K = 24$ lb/rad., and at the same time, there seems to be only a slight improvement in the phugoid response. (Compare Figures III-19 and III-26.)

The preceding figures were for a pilot-basic airframe combination. The effects of augmentation on the system will now be investigated. For illustrative purposes, an airframe with a tuck-under condition will be used. In addition, the short period mode of the basic airframe is poorly damped, requiring approximately 5 cycles to damp to a small value. Stability derivatives are given in Table III-3. The time history of the airframe in this condition is shown in Figure III-28. The diverging phugoid and poorly damped short period are clearly visible in the traces.

$U_0 = 915$	$X_u = -.126$
$K_{\delta} = 160$	$X_w = -.0296$
$M_u = -.0164$	$Z_u = -.0743$
$M_w = -.02$	$Z_w = -3.13$
$M_{\dot{w}} = .00124$	$Z_{\delta} = -19$
$M_{\dot{\theta}} = 1.5$	$Z_{\delta_s} = 53.6$
$M_{\delta_s} = 20.5$	

Table III-3. Airframe Parameters Used in Figures III-28 through III-50

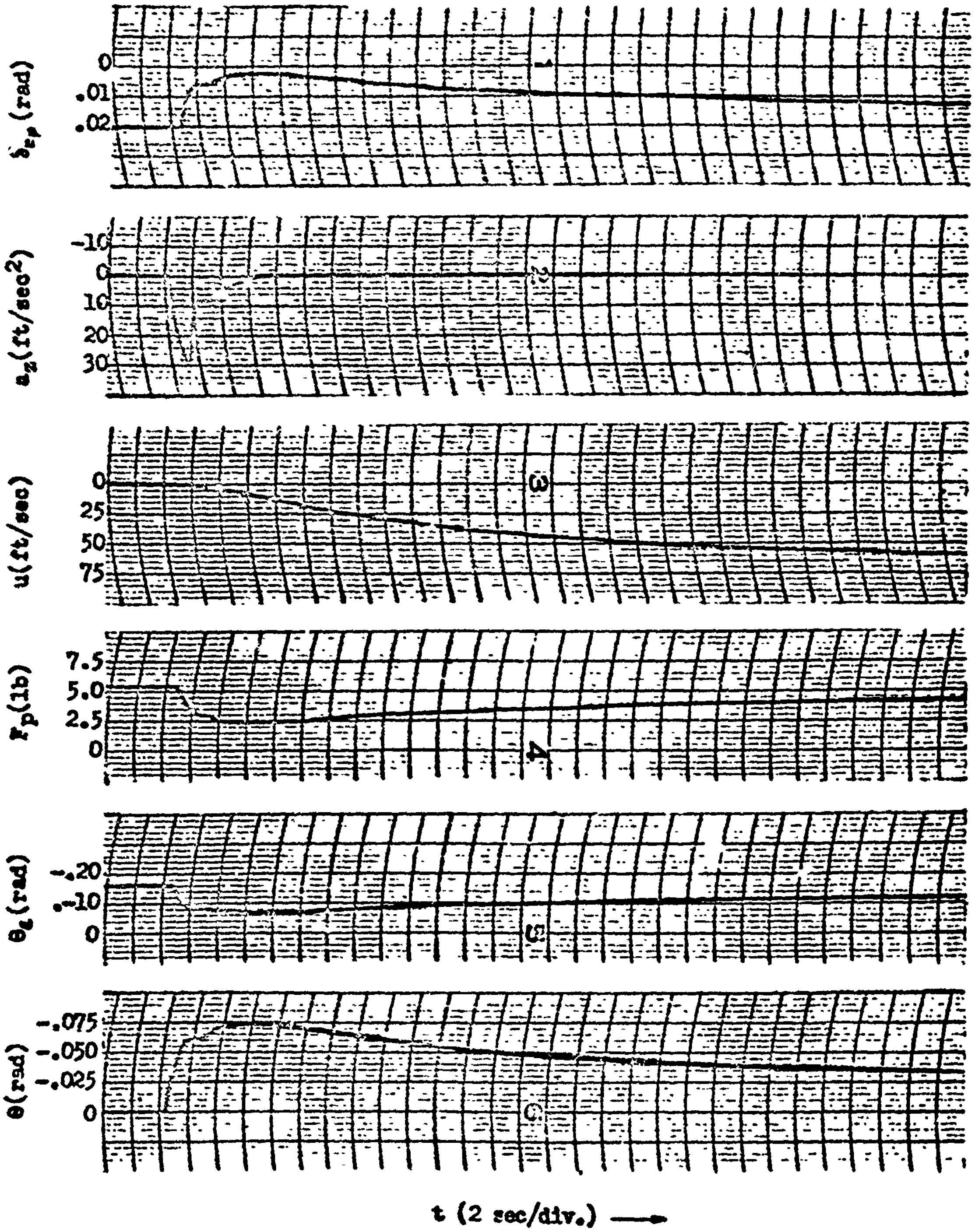


Figure III-24. Response of Pilot-Unaugmented Airframe System to Step θ_{ref} Command for Same Conditions as in Figure III-19 except K= 40 lb/rad.

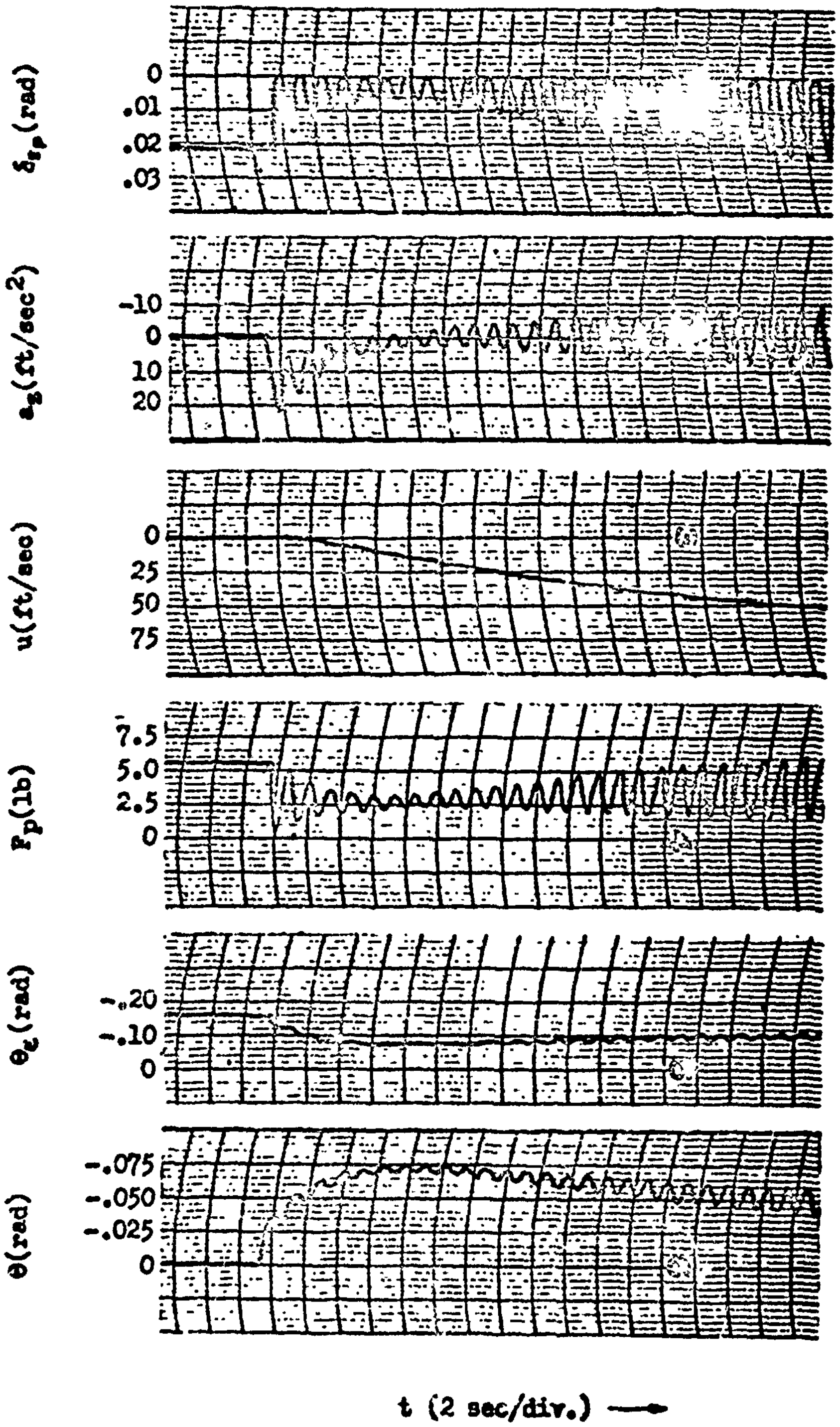


Figure III-25. Response of Pilot-Unaugmented Airframe System to Step θ_{ref} Command for Same Conditions as Figure III-24 except T_I = 2 sec.

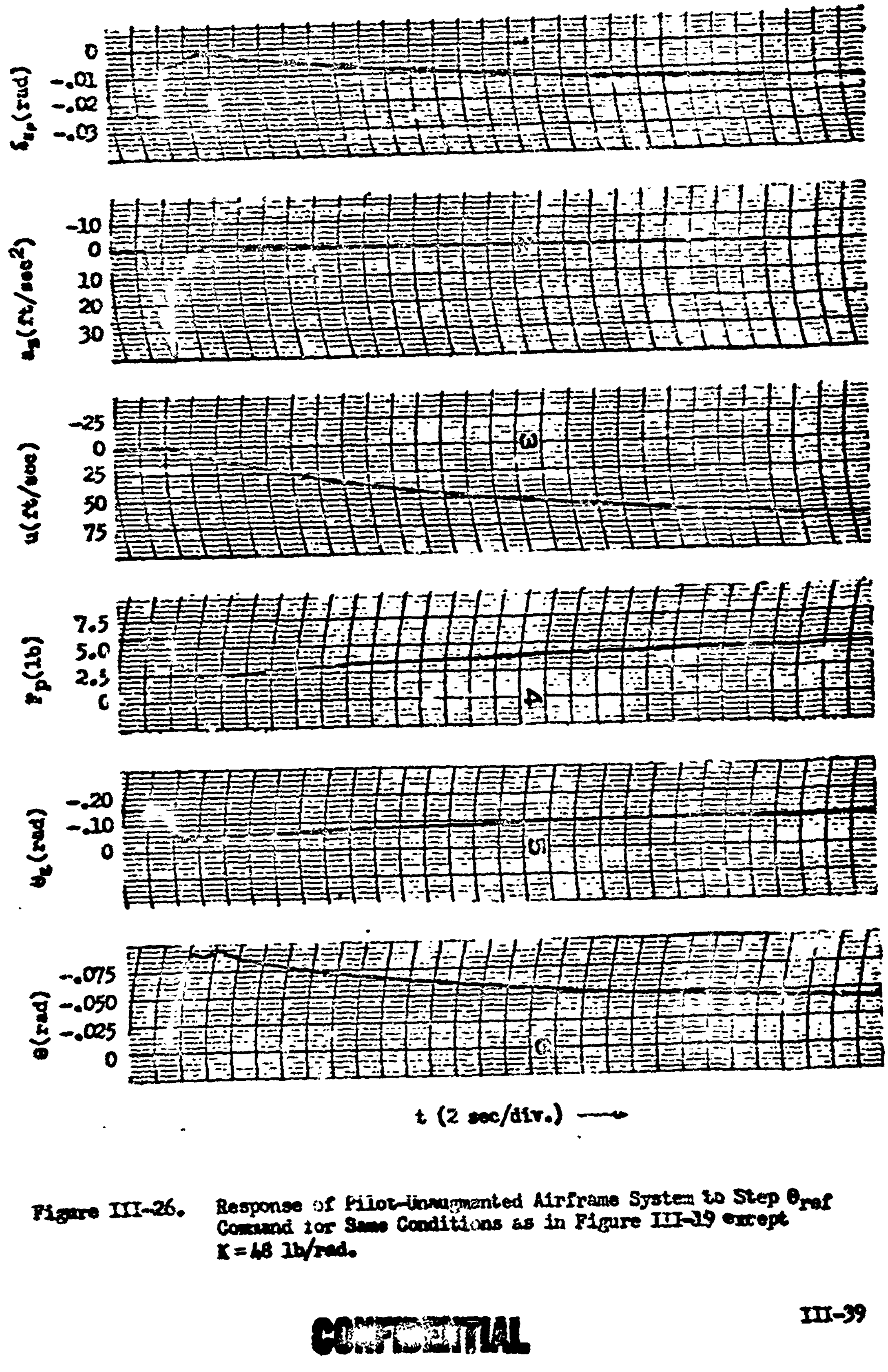


Figure III-26. Response of Pilot-Unaugmented Airframe System to Step θ_{ref} Command for Same Conditions as in Figure III-19 except K = 48 lb/rad.

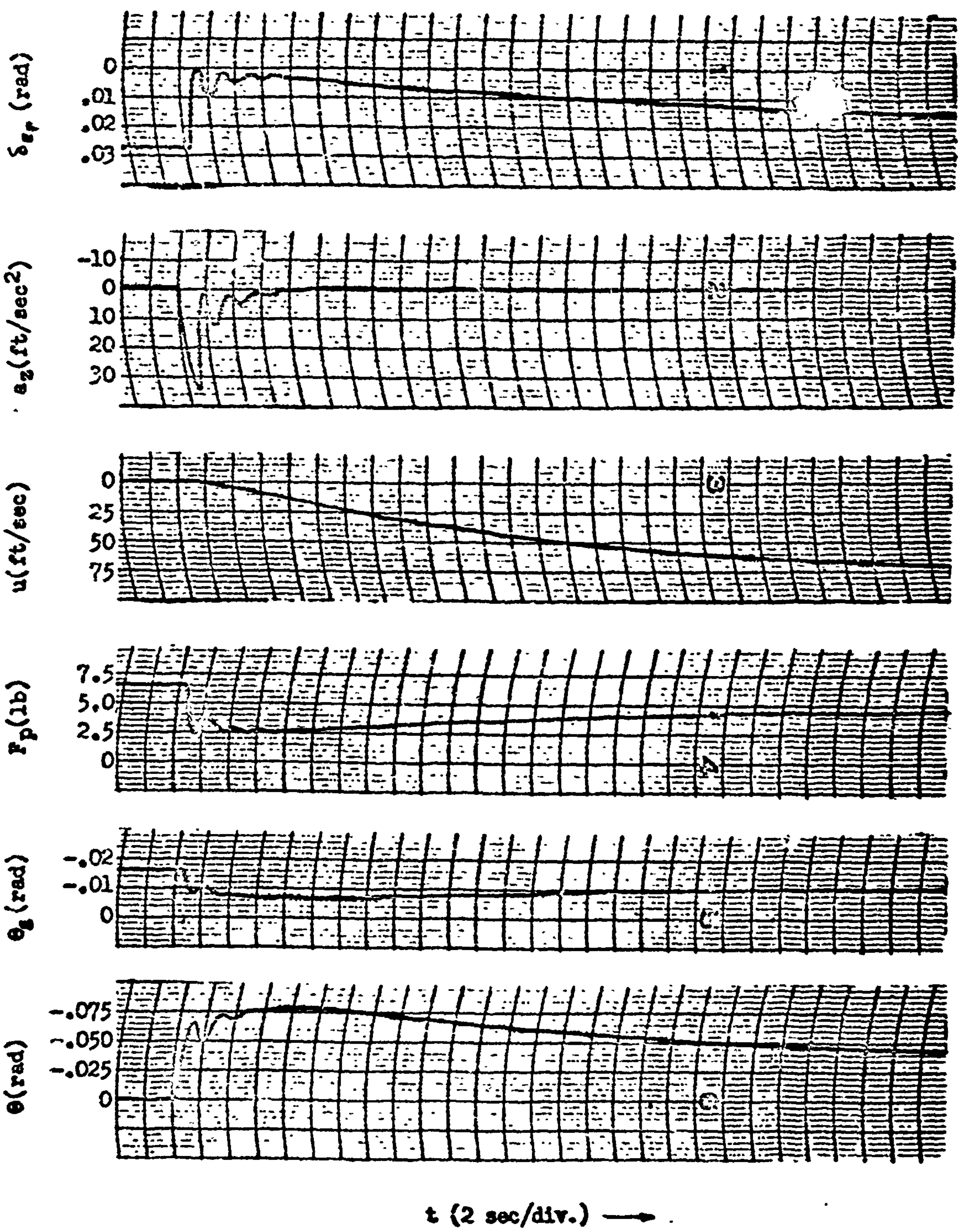


Figure III-27. Response of Pilot-Unaugmented Airframe System to Step θ_{ref} Command for Same Conditions as in Figure III-26 except $T_1 = 1$ sec.

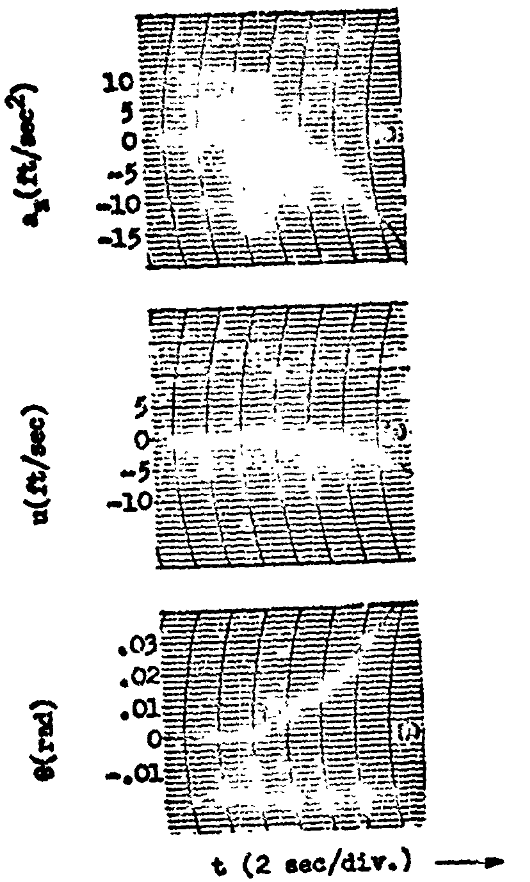


Figure III-28. Response of Unaugmented Airframe to θ Gust Input for Conditions of Table III-3.

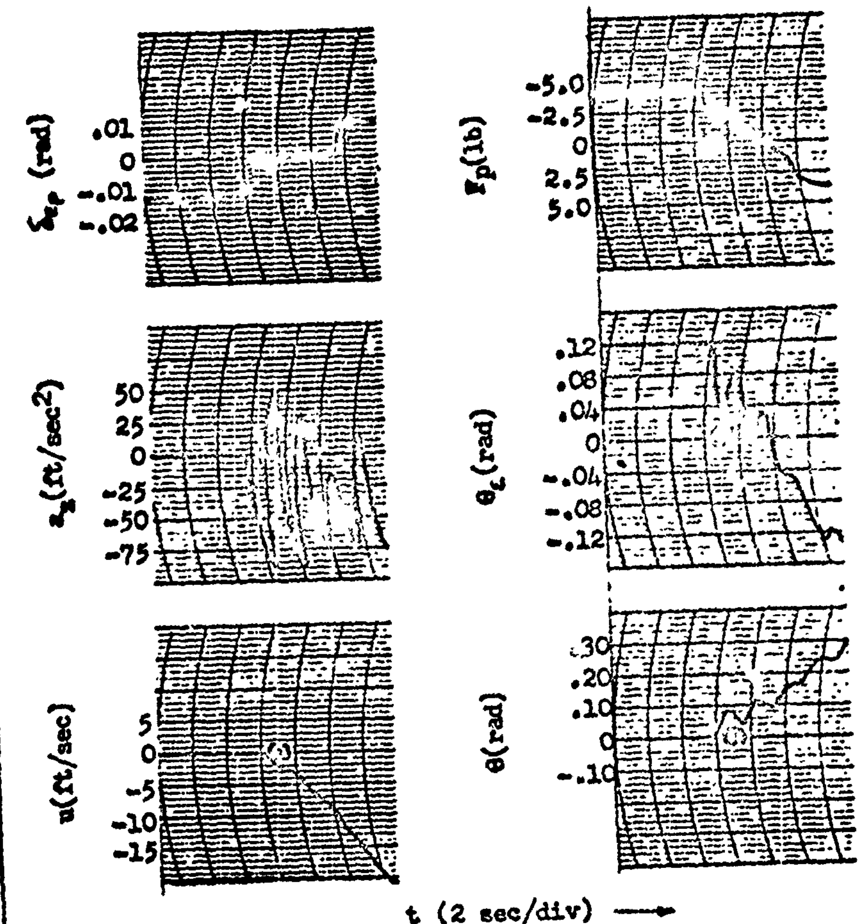


Figure III-29. Response of Pilot-Unaugmented Airframe to Step θ_{ref} Command for Conditions of Table III-3 ($K = 32$ lb/rad, $T = 0.3$, $T_1 = 0$, $T_2 = 0.2$ sec).

Figure III-29 indicates the difficulty which the pilot encounters in trying to control such an airplane. The short period oscillations are still pronounced and the diverging phugoid cannot be controlled.

Consider first the phugoid divergence. (III-28) showed that by using u augmentation, the tuck-under tendency can be eliminated. This is shown in Figures III-30, III-31, and III-32. In Figure III-30, the amount of u augmentation is given by $Z_u' = .040$. Then

$$\text{(III-45)} \quad \left[\begin{aligned} Z_{u_T} &= Z_u + Z_u' = -.0743 + .040 = -.0343 \\ \ddot{M}_{u_T} &= M_u + M_u' \\ &= M_u + KZ_u' = -.0164 + (.38)(.040) = -.0012 \end{aligned}\right.$$

and

$$\text{(III-46)} \quad \begin{aligned} E' &= g(Z_{u_T} M_w - M_{u_T} Z_w) \\ &= g[(-.0343)(-.02) - (-.0012)(-3.13)] \\ &= -.0031g \end{aligned}$$

Since E' is still negative, the phugoid should still diverge. This can be seen in the figure.

In Figure III-31, $Z_u' = .045$

and

$$\text{(III-47)} \quad E' = g[(-.0393)(-.02) - (+.0007)(-3.13)] = +.003g$$

Thus for this case, the phugoid is stabilized, as shown in Figure III-31. In Figure III-32, the value of Z_u' has been increased to .050 with no noticeable improvement in the phugoid response.

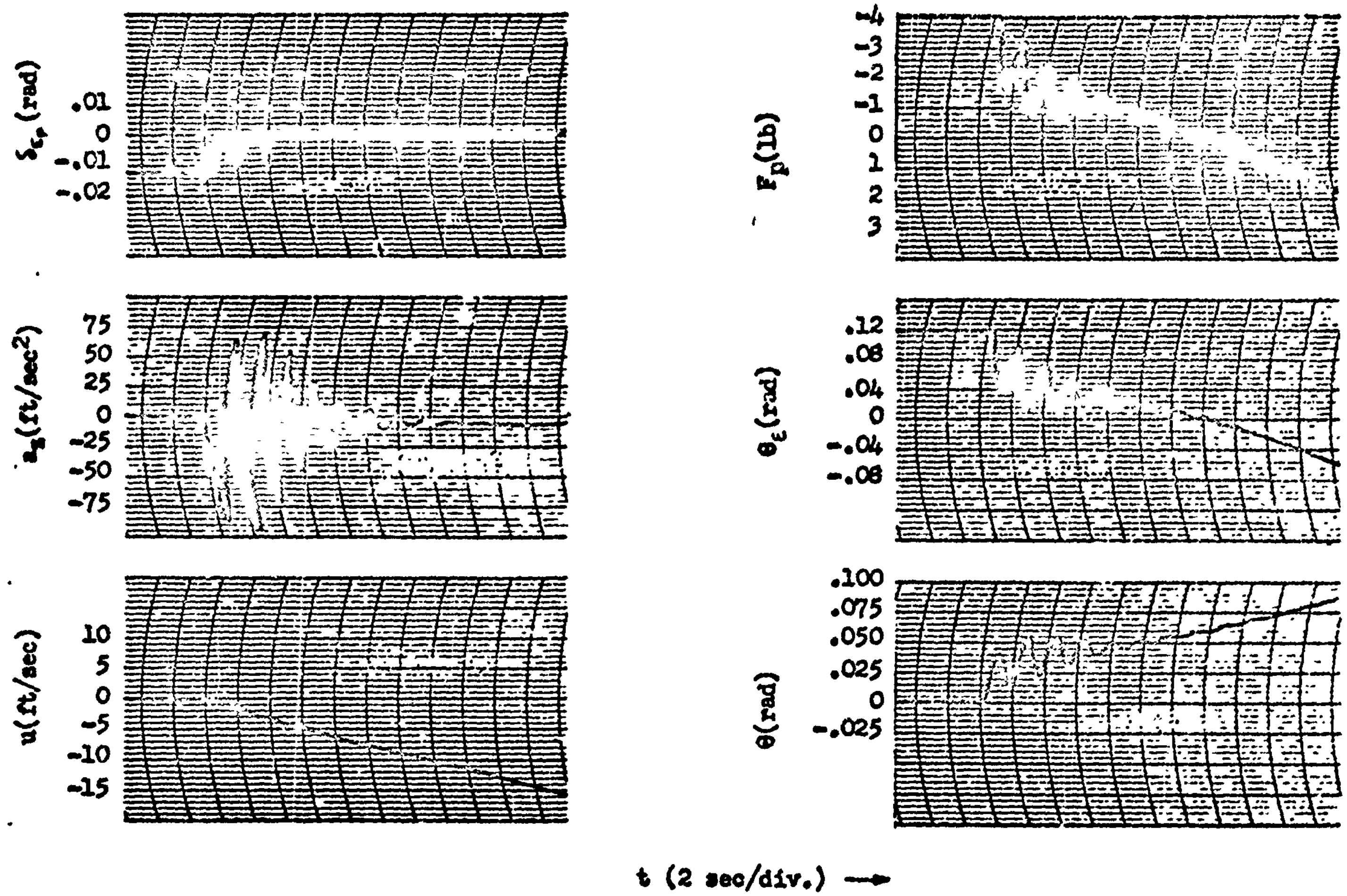


Figure III-30. Response of Pilot-Airframe System to Step θ_{ref} Command for Conditions of Figure III-29 and with u Augmentation ($Z_u' = 0.040$).

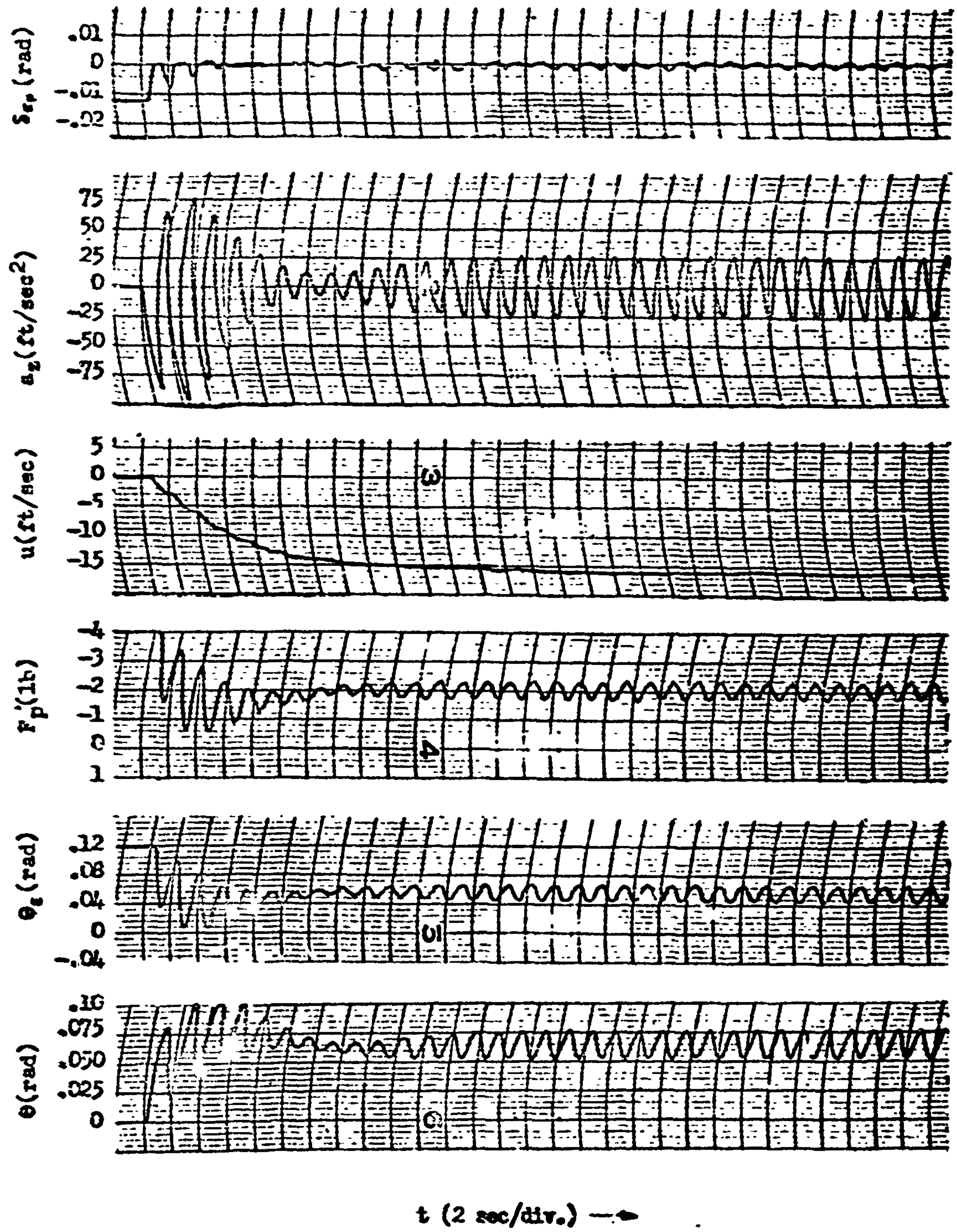


Figure III-31. Response of Pilot-Airframe System to Step θ_{ref} Command for Conditions of Figure III-29 and with u Augmentation ($Z_u' = 0.045$).

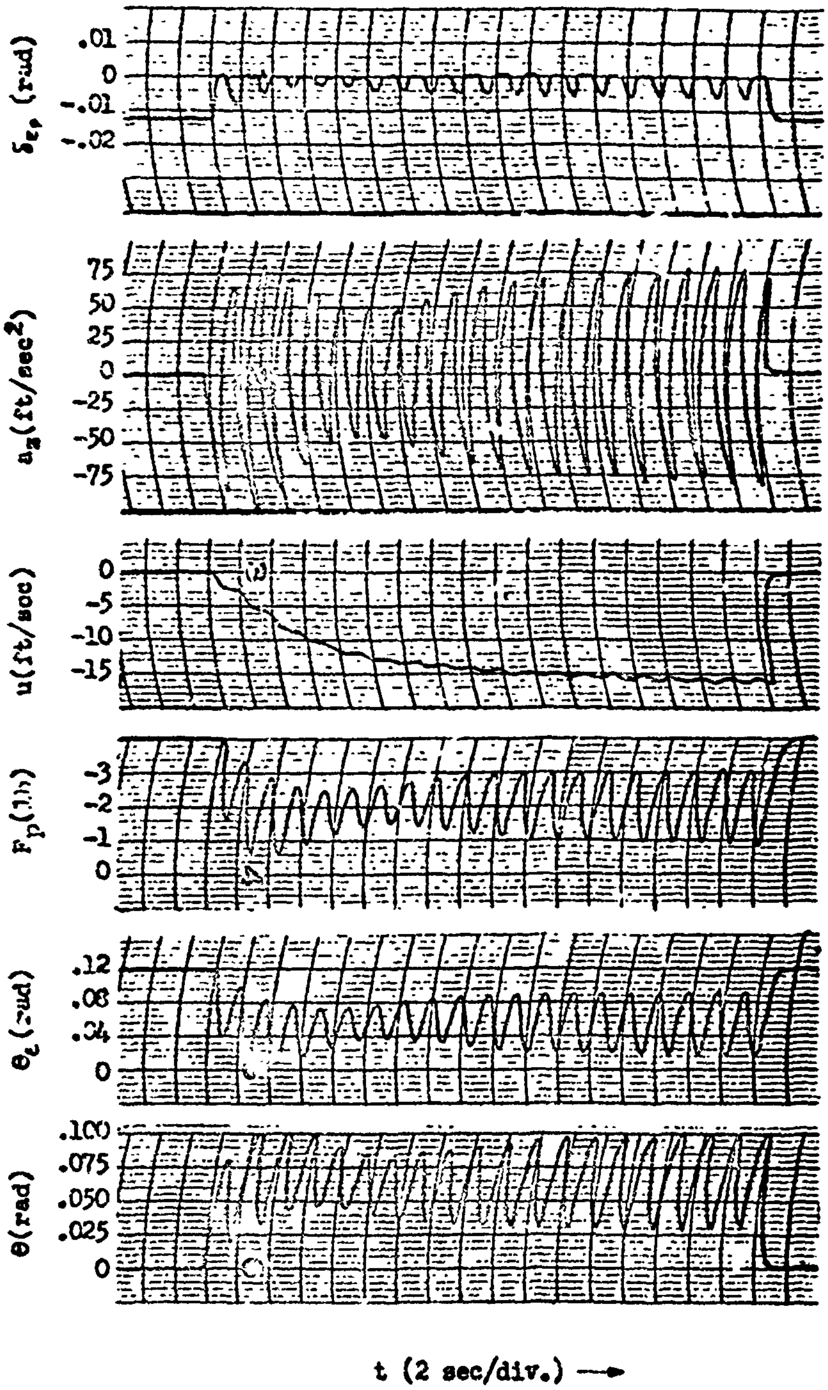


Figure III-32. Response of Pilot-Airframe System to Step θ_{ref} Command for Conditions of Figure III-29 and with u Augmentation ($Z_u' = 0.050$).

Although the phugoid has been stabilized by u augmentation, the high frequency oscillations are still evident with the damping ratio decreasing as the u augmentation is increased. The high frequency oscillations are damped out by using $\dot{a}_z$ augmentation as shown in Figure III-33. The short period oscillations have been eliminated, and for $Z_{\dot{a}_z} = .0050$ the system response is rapid and has no large overshoots. For any value of $Z_{\dot{a}_z}$ smaller than the one given, the short period oscillations persist for a few cycles, the number of cycles depending on the value of $Z_{\dot{a}_z}$.

Figure III-34 shows the effect of a_z feedback. Note that the height of the a_z peak decreases as Z_{a_z} is increased while the magnitude of the applied force stays constant. This would tend to indicate that the stick force per g increases as the a_z feedback is increased.

This fact is more evident when the two degree of freedom short period equations are examined. These equations are

$$\text{(III-48)*} \quad \left[\begin{aligned} &\dot{w} - U_0\dot{\theta} = Z_w w + Z_{\delta_z}\delta_z = a_z \\ &\ddot{\theta} = M_w w + M_{\dot{w}}\dot{w} + M_{\dot{\theta}}\dot{\theta} + M_{\delta_z}\delta_z \end{aligned}\right.$$

From (III-48)

$$\text{(III-49)} \quad \frac{a_z}{\delta_z} = \frac{\dot{w} - U_0\dot{\theta}}{\delta_z} = \frac{Z_{\delta_z}(s^2 + a_1 s + a_0)}{(s^2 + 2\zeta\omega_n s + \omega_n^2)}$$

where

$$a_1 = -(M_{\dot{\theta}} + U_0 M_{\dot{w}})$$

$$a_0 = -(U_0/Z_{\delta_z})(M_w Z_{\delta_z} - M_{\delta_z} Z_w) < 0$$

$$\omega_n = (Z_w M_{\dot{\theta}} - U_0 M_w)^{\frac{1}{2}}$$

$$\zeta = -(Z_w + U_0 M_{\dot{w}} + M_{\dot{\theta}})/2\omega_n$$

* See Reference 4, (III-24) and (III-10).

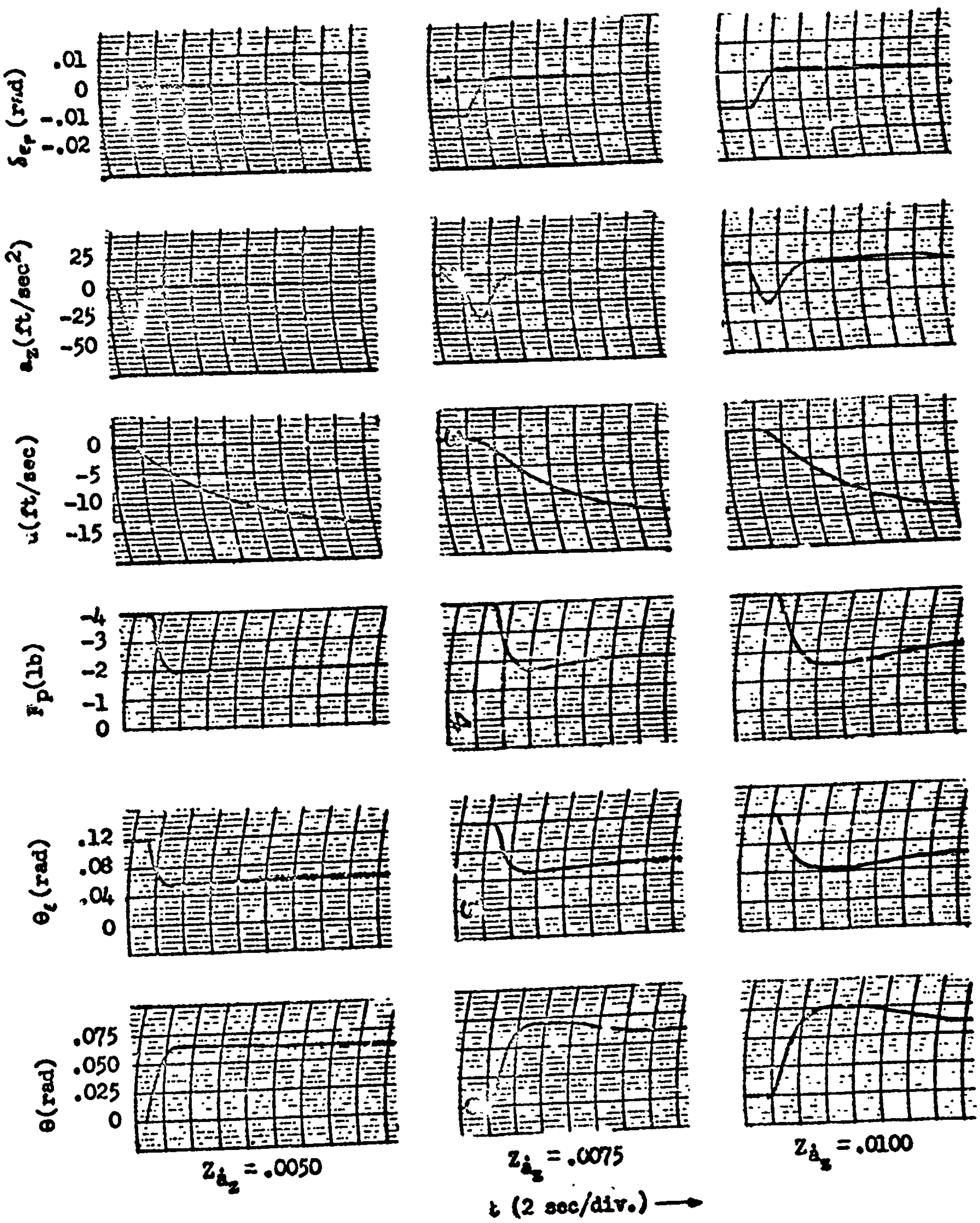


Figure III-33. Response of Pilot-Airframe System to Step θ_{ref} Command for Conditions of Figure III-29 with u and $\dot{a}_z$ Augmentation ($Z_u' = .045$).

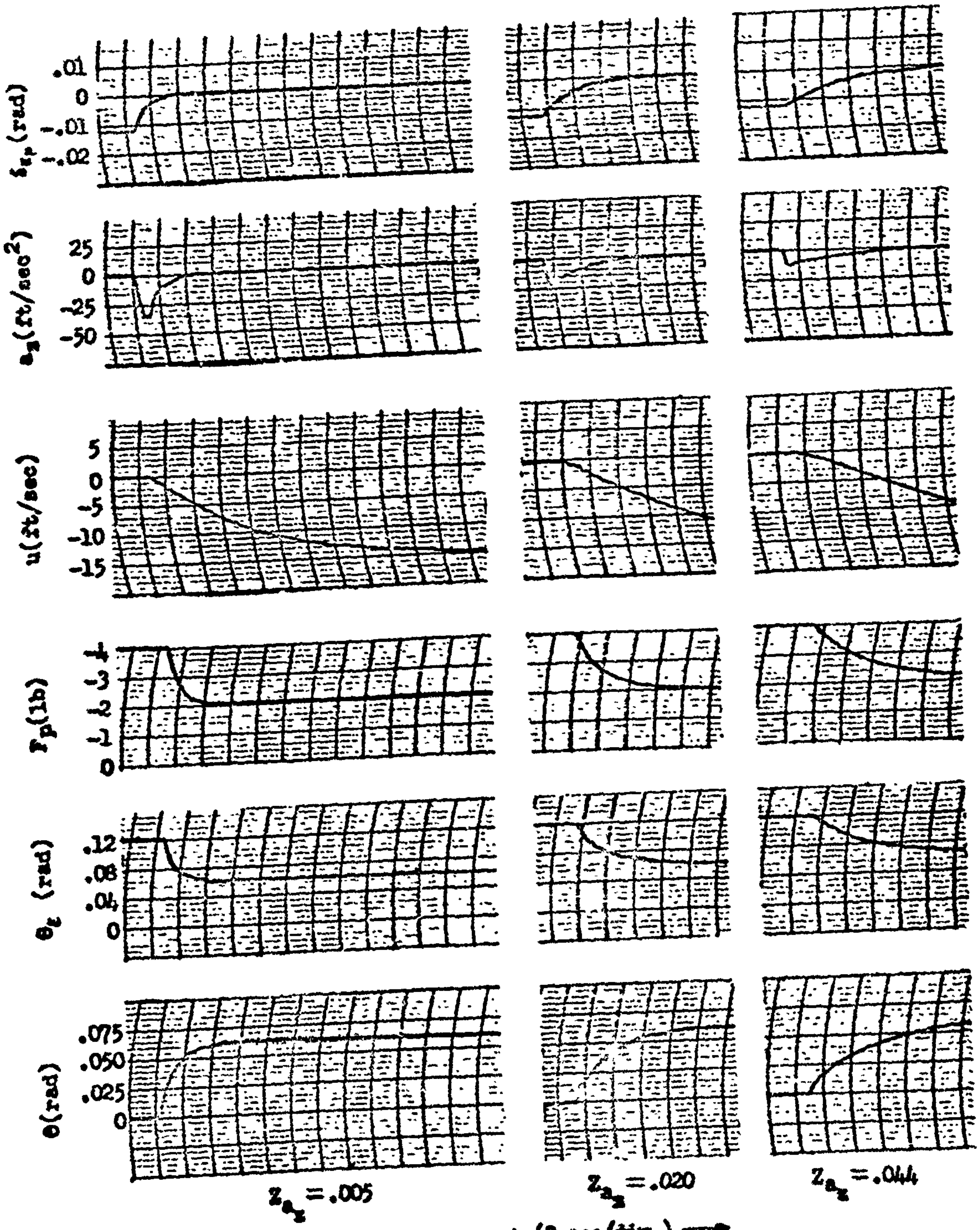


Figure III-34. Response of Pilot-Airframe System to Step θ_{ref} Command for Conditions of Figure III-29 with u, $\dot{a}_z$, and a_z Augmentation ($Z_u' = .045$, $Z_{\dot{a}_z} = .005$).

Noting that the change in load factor in g unit is

(III-50) $\Delta n = -\frac{a_z}{g}$

and that the total elevator deflection (assuming only a_z feedback) is

(III-51) $\delta_e = \delta_{F_p} + K_{a_z} a_z = \frac{F_p}{K_{\delta_e}} - g K_{a_z} \Delta n$

then

(III-52) $$\frac{\Delta n}{(F_p / K_{\delta_e}) - g K_{a_z} \Delta n} = -\frac{Z_{\delta_e}}{g}\left(\frac{s^2 + a_1 s + a_0}{s^2 + 2\zeta\omega_n s + \omega_n^2}\right)$$

or

(III-53) $$\Delta n = \frac{Z_{\delta_e}(s^2 + a_1 s + a_0)}{g\left[(1 - K_{a_z} Z_{\delta_e})s^2 + (2\zeta\omega_n - K_{a_z} Z_{\delta_e} a_1)s + (\omega_n^2 - K_{a_z} Z_{\delta_e} a_0)\right]} \frac{F_p}{K_{\delta_e}}$$

Letting F_p be a step input, i.e., $F_p = |F_p|/K_{a_z}$, the steady state value of Δn is

(III-54) $$\Delta n_{ss} = \frac{Z_{\delta_e} a_0}{g(K_{a_z} Z_{\delta_e} a_0 - \omega_n^2)} \frac{|F_p|}{K_{\delta_e}}$$

or

(III-55) $$\frac{|F_p|}{\Delta n} = \frac{g K_{\delta_e}(K_{a_z} Z_{\delta_e} a_0 - \omega_n^2)}{Z_{\delta_e} a_0}$$

For the condition chosen,

$$Z_{\delta_e} a_0 = -58,692$$

$$\omega_n^2 = 13.6$$

Consider first the case where $K_{\dot{\theta}_E} = 0$; then

$$\frac{|F_p|}{\Delta n} = \frac{(32.2)(160)(-13.6)}{(-58{,}692)} = 1.19 \text{ lb/g}$$

For the case where $K_{\dot{\theta}_E} = .0000932$,

$$\frac{|F_p|}{\Delta n} = \frac{(32.2)(160)[(.0000932)(-58{,}692) - 13.6]}{-58{,}692} = 1.67 \text{ lb/g}$$

Note that while the slope of the curve of the stick force versus change in load factor increases as the amount of $\dot{\theta}_E$ feedback is increased, the system response time also increases.

An alternate method of raising the stick force per g gradient without changing the system response is evident in (III-55). This equation indicates that $K_{\delta_{E}}$, the control system spring constant, can be increased to raise $|F_p|/\Delta n$. If, at the same time, the pilot gain K is increased to keep the ratio of K to K_{δ_E} constant, the system response should remain unchanged since the ratio of δ_{E_p} to θ_E will remain unchanged, as shown in (III-40a).

It is interesting to note that $\dot{u}$ augmentation is not required for this case, due to the fact that as long as the phugoid is not exponentially diverging, the pilot can damp out these long period oscillations. Of course, the effort required to do this should not become extreme. Examination of the pilot's force curve in Figure III-34 shows that the pilot is not exerting much effort to reach a steady state value without overshooting or hunting. This fact is more evident when Figures III-30 and III-34 are compared.

Augmentation has improved the pilot's control over the airframe to a great extent, but what has it done to the airframe itself? The effect of augmentation on the basic airframe is presented in Figure III-35. These time histories correspond to a condition where the pilot has his hands off the control stick and the airframe is disturbed by a vertical gust of wind. The contrast between

Figures III-35 and III-28 shows vividly the stabilizing influence of augmentation. The diverging phugoid mode has been stabilized and the short period damping ratio has been increased to a point where the short period oscillation disappears within one cycle.

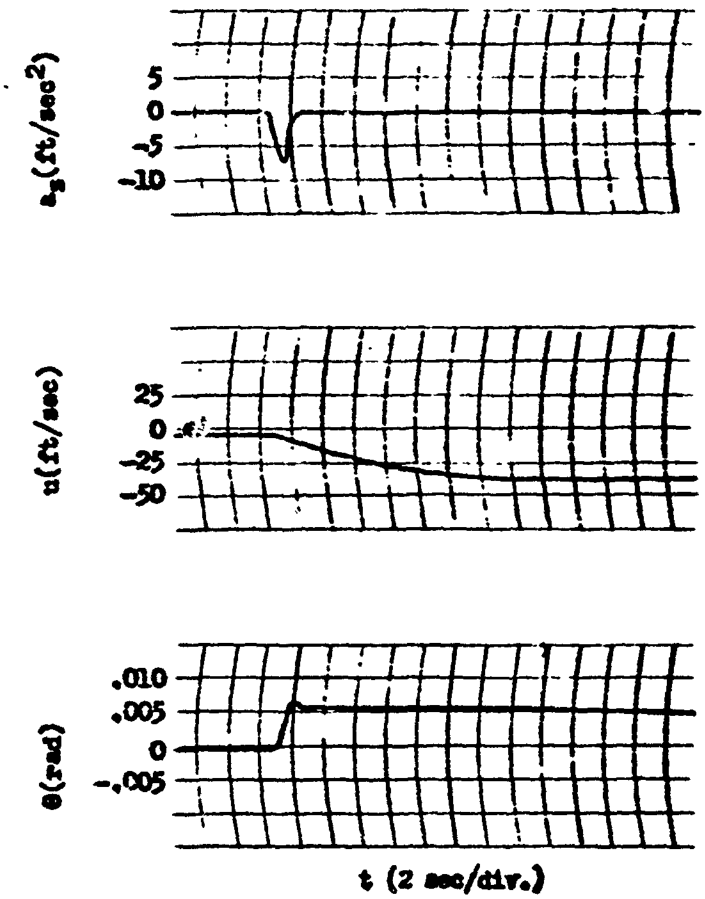


Figure III-35. Response of Augmented Airframe to θ Gust Input ($Z_u' = .045$, $Z_{\dot{a}_z} = .005$, $Z_{a_z} = .005$)

The question now arises: what happens to this optimized system when the pilot parameters vary due to fatigue, tenseness, or carelessness? The effects of pilot parameter variation are shown in Figures III-36 through III-50.

Figures III-36 through III-42 show the trend as the reaction time, τ, is varied from .1 second to 1.5 seconds.* The normal variations of τ for aircraft pilots is expected to be from .25 second to .8 second. It can be seen that the reaction time variation has no effect on the system stability but does influence the system response.

Consider Figure III-42, where $\tau = 1.5$ seconds. The pilot sees the error θ_ε building up from zero and realizes that he should exert a certain force to bring this error down. However, he does not react until 1.5 seconds has elapsed. By this time, θ and θ_ε have increased to sizable values. When the pilot's control finally becomes effective, θ and θ_ε gradually decrease. In the end, θ_ε for both $\tau = 1.5$ and $\tau = 0.3$ will be the same, but initially, there will be a larger θ_ε for $\tau = 1.5$ than for $\tau = 0.3$. Since variations in τ affect mainly the initial error and do not affect system stability, it can be concluded that optimum augmentation need not concern itself with the value of τ.

Figures III-43 and III-44 illustrate again the fact that increasing the rate judgment time constant T_1 introduces high frequency oscillations. Of course, these short period oscillations can be damped out by using more $\dot{\delta}_e$ augmentation. Since most pilots fly by rate as well as by displacement, it would perhaps be more realistic to augment the system with a certain amount of rate judgment included in the human pilot transfer function.**

Figure III-45 is for the same conditions as in Figure III-34 except that the pilot's neuro-muscular lag time constant T_2 has been increased from .2 to 1 second. Although the time constant has been increased by a factor of five,

* For a step θ_{ref} input, variation of τ will vary the dead-time between signal perception and response with little effect on the shape of the response curves. Therefore, for a study of the effects of τ, the error signal θ_ε will be initiated when θ, the airframe pitch attitude, is disturbed from θ_{ref}, where θ_{ref} is zero. The pilot then tends to control the airplane to bring θ back to zero. Here again, it will be noted that θ_ε never returns to zero.

** The previous augmentation was done with zero rate judgment.

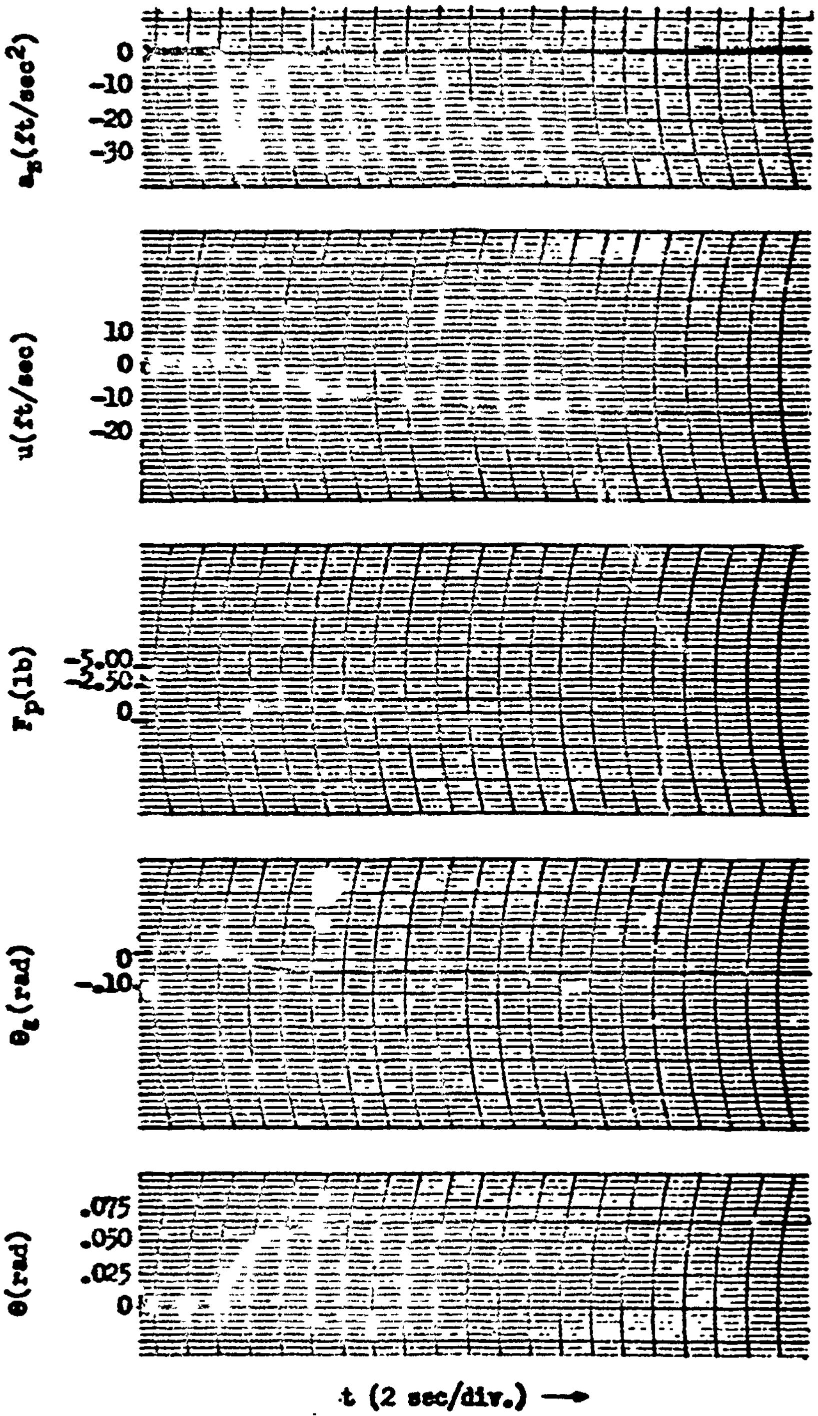


Figure III-36. Response of Pilot-Augmented Airframe System to θ Disturbance for Conditions of Figure III-29 except τ = 0.1 sec.

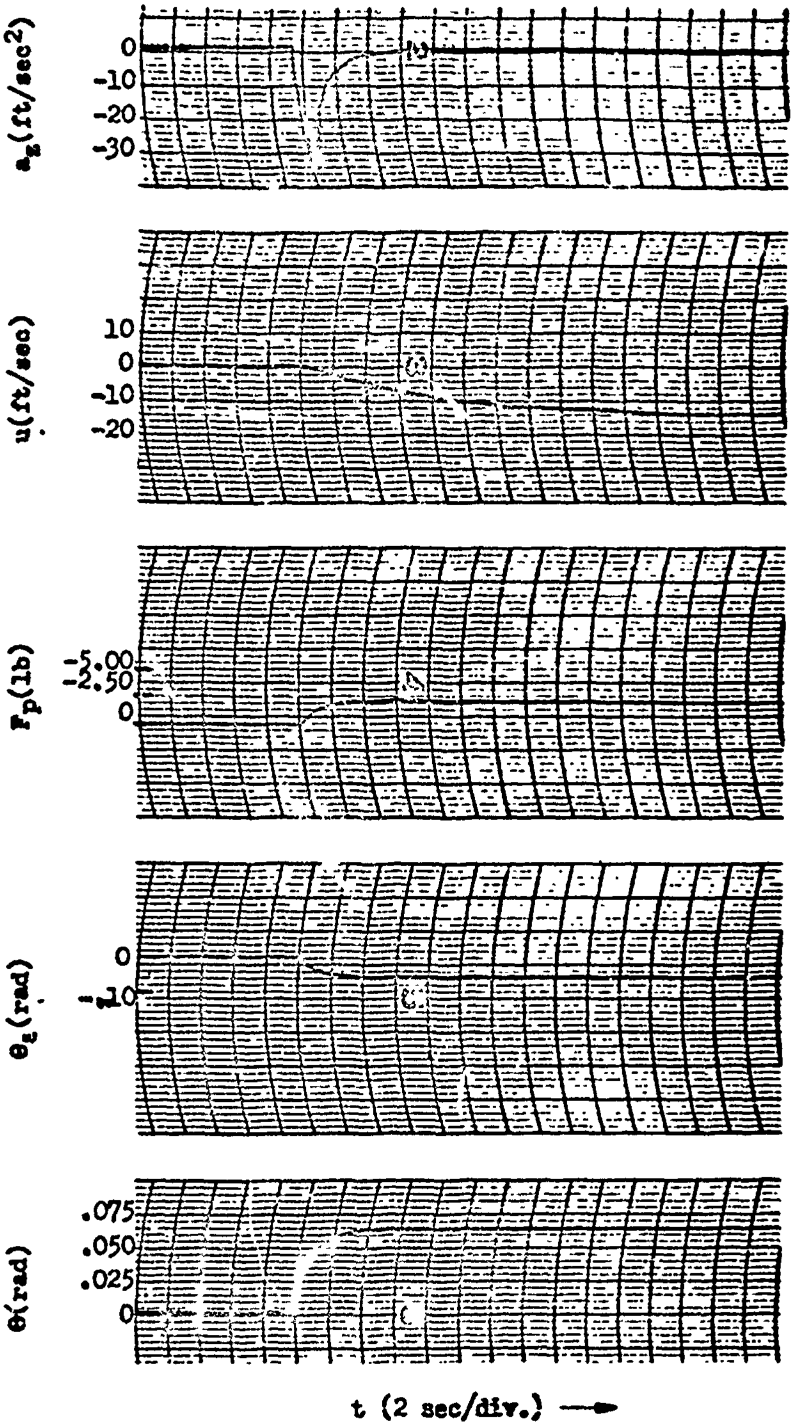


Figure III-37. Response of Pilot-Augmented Airframe System to θ Disturbance for Conditions of Figure III-29 except $\tau = 0.3$ sec.

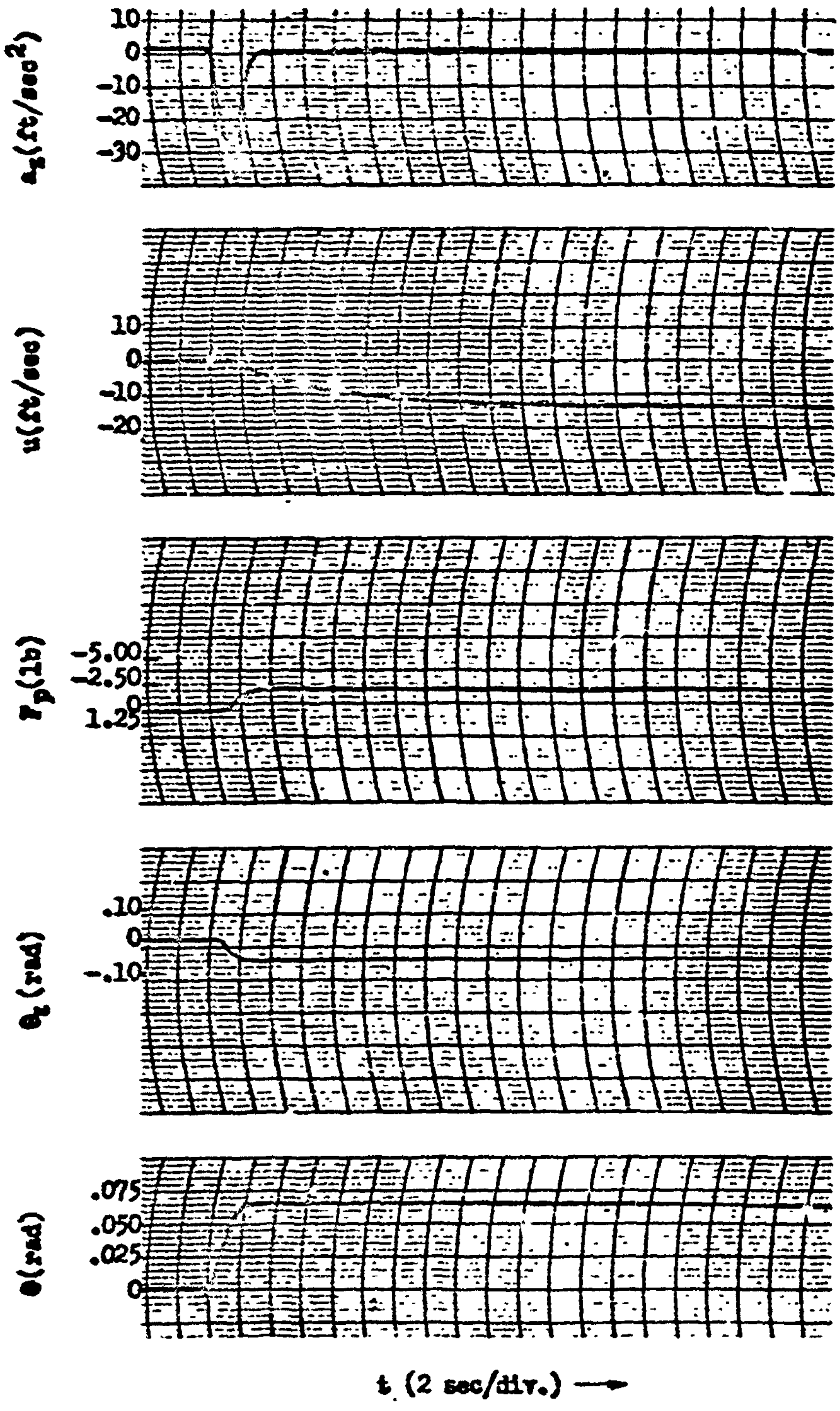


Figure III-38. Response of Pilot-Augmented Airframe System to G Disturbance for Conditions of Figure III-29 except $\tau = 0.6$ sec.

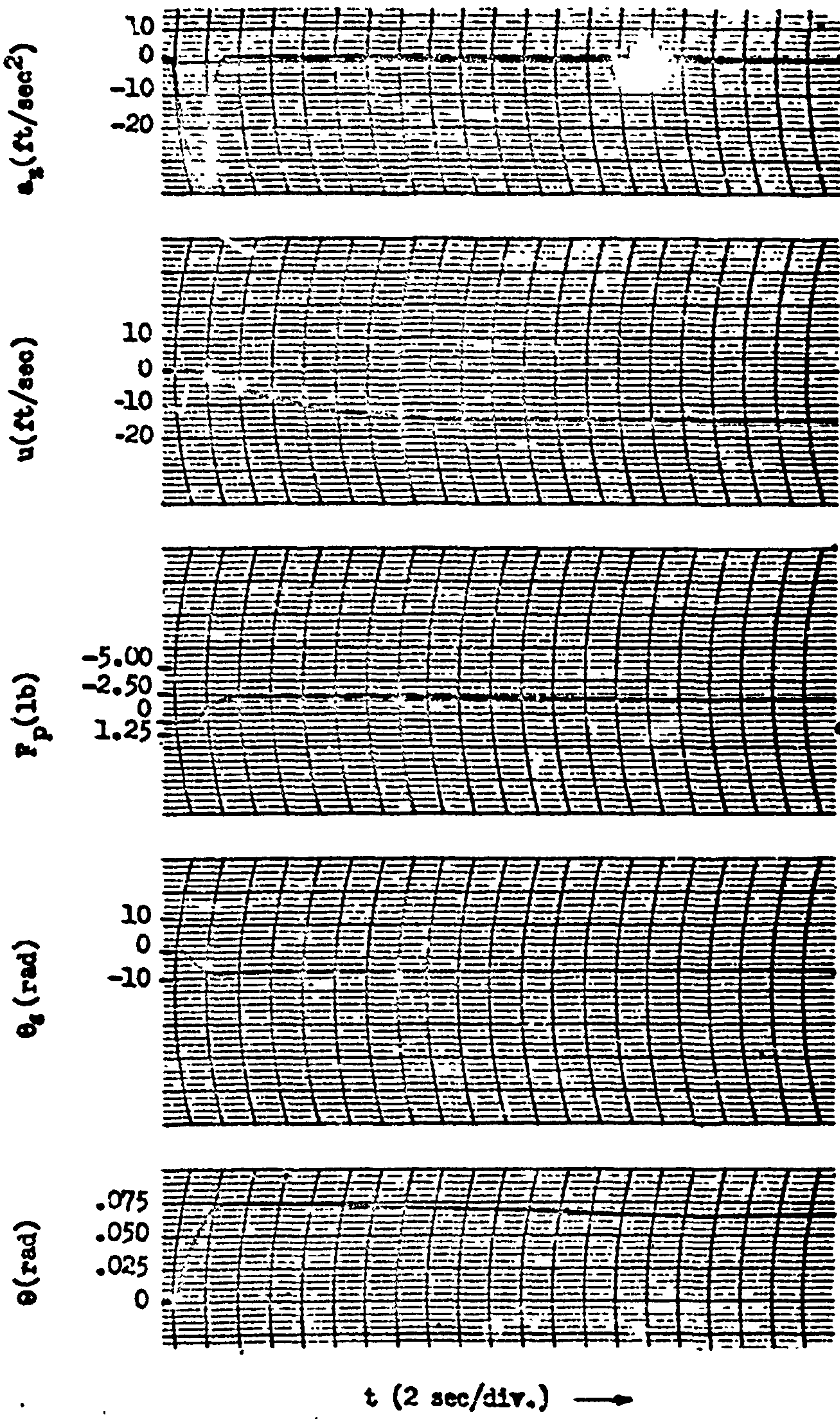


Figure III-39. Response of Pilot-Augmented Airframe System to θ Disturbance for Conditions of Figure III-29 except $\tau = 0.9$ sec.

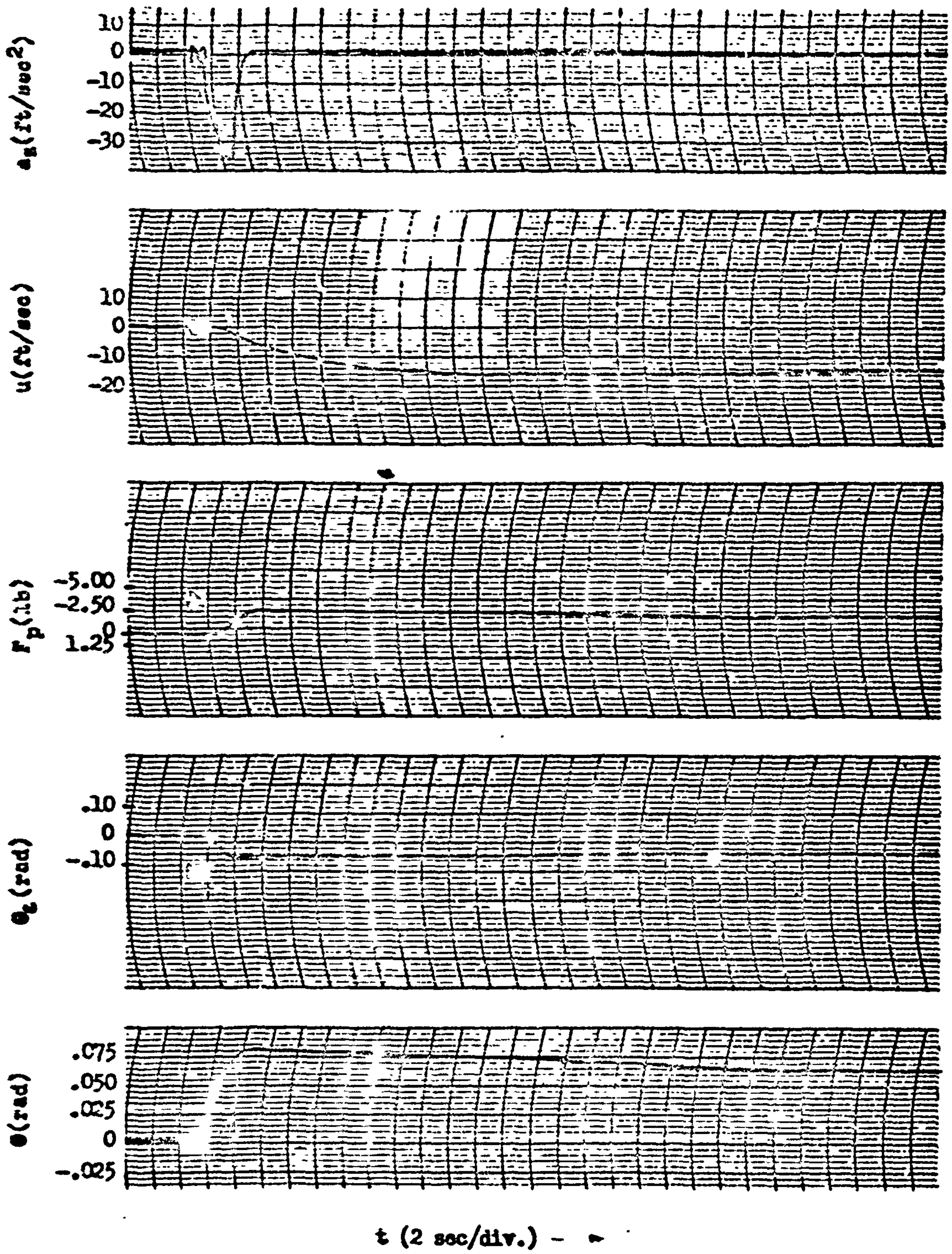


Figure III-40. Response of Pilot-Augmented Airframe System to θ Disturbance for Conditions of Figure III-39 except τ=1.0 sec.

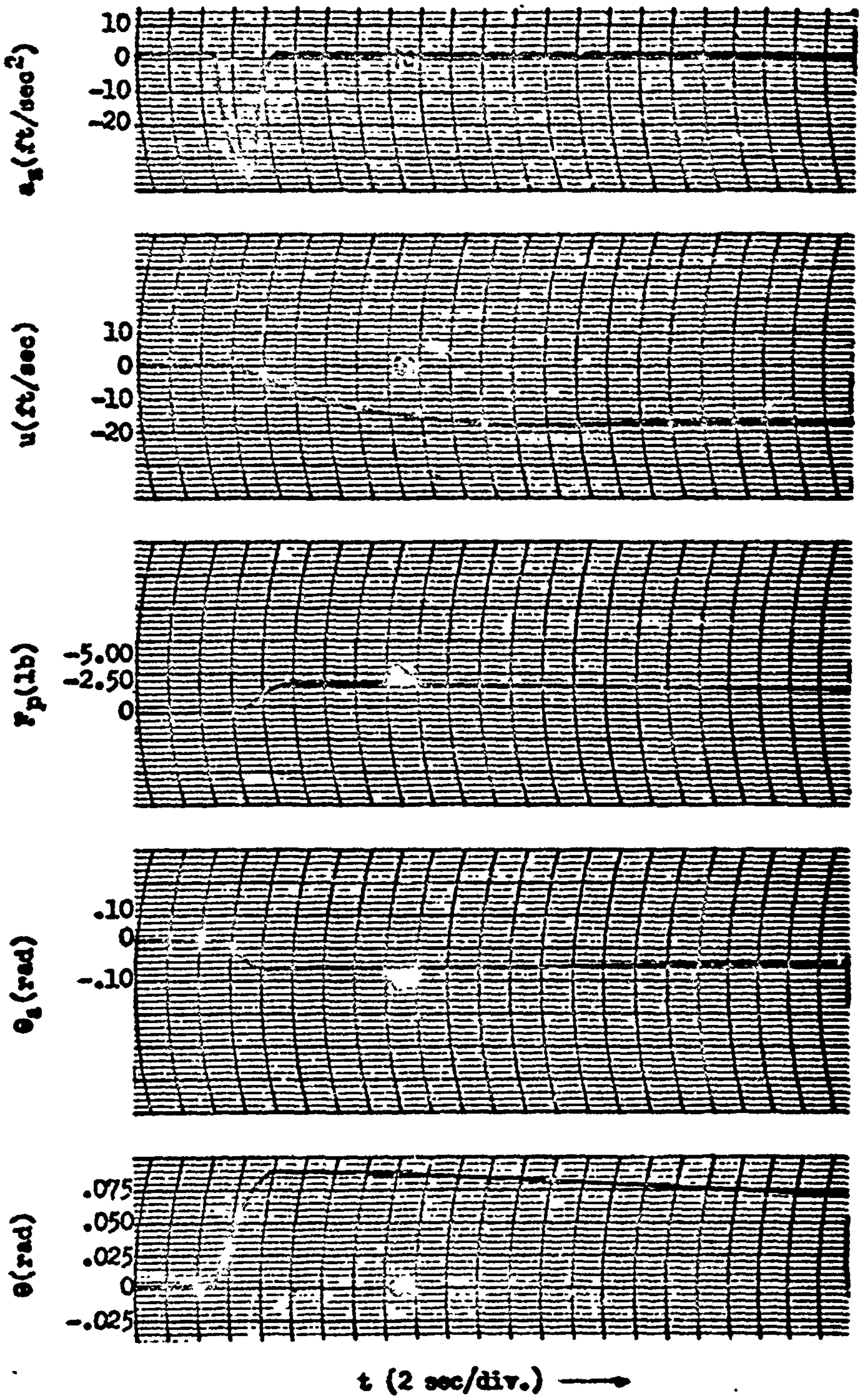


Figure III-41. Response of Pilot-Augmented Airframe System to θ Disturbance for Conditions of Figure III-29 except T=1.25 sec.

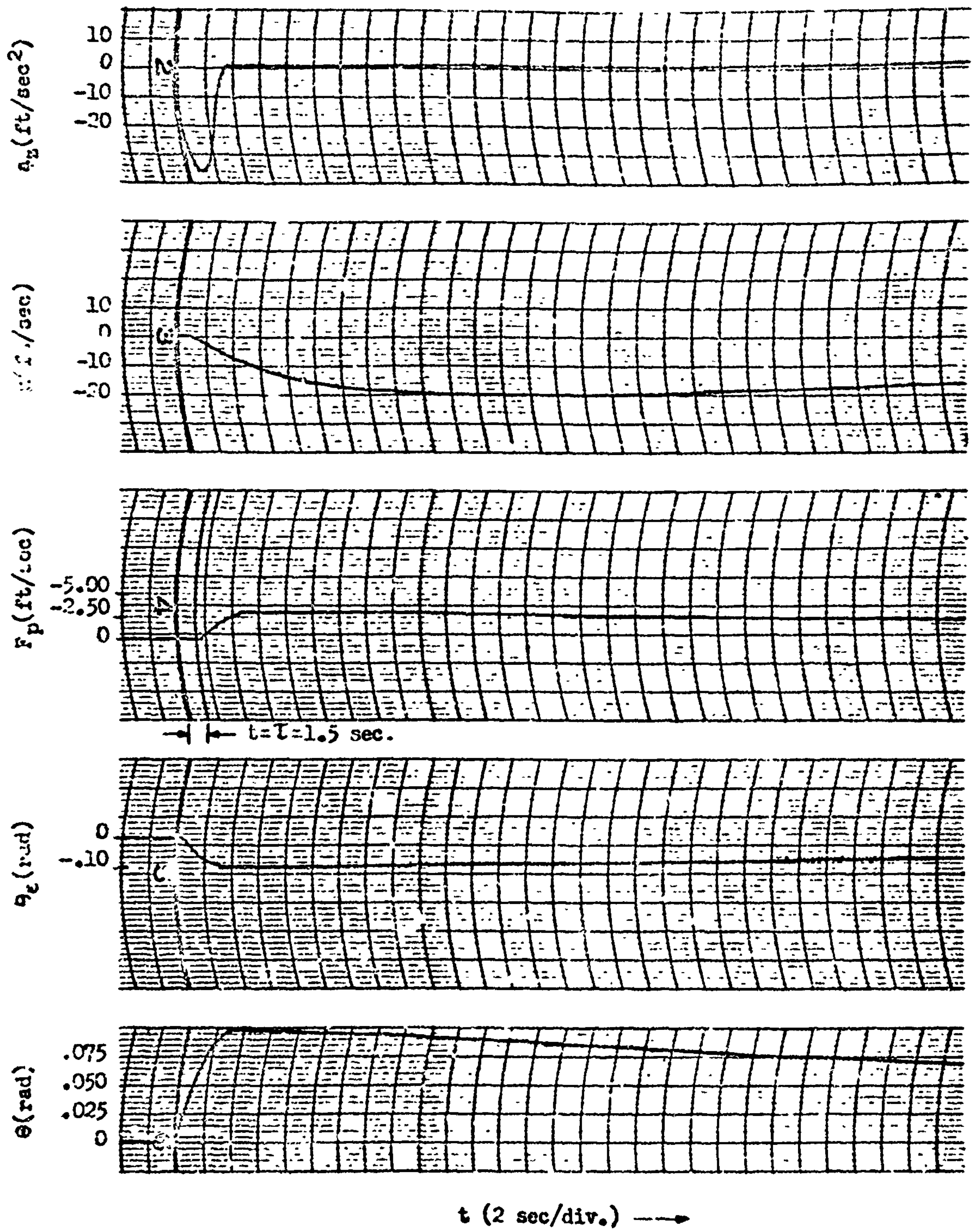


Figure III-42. Response of Pilot-Augmented Airframe System to θ Disturbance for Conditions of Figure III-29 except τ = 1.5 sec.

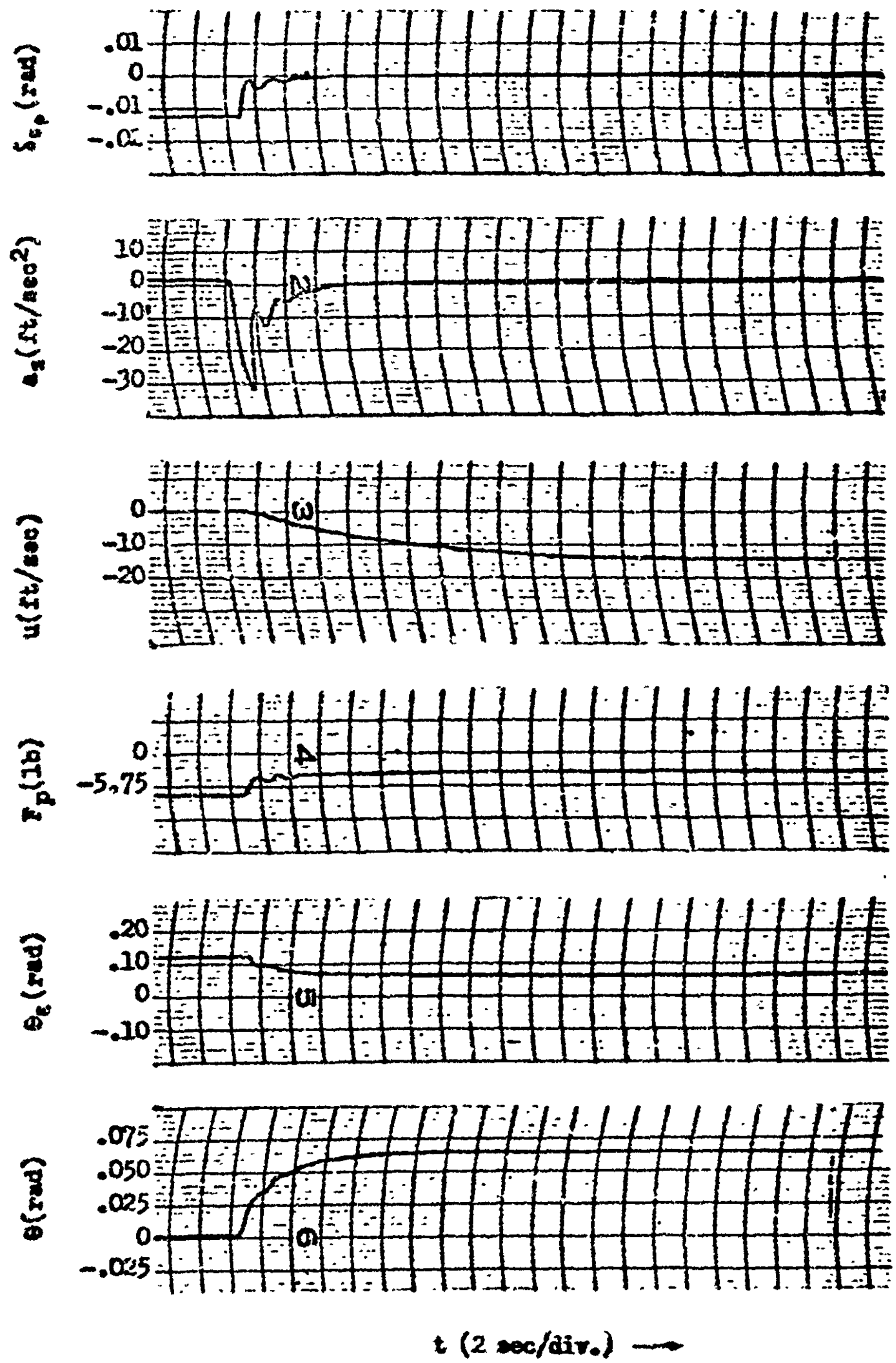


Figure III-43. Response of Pilot-Augmented Airframe System to Step θ_{ref} Command for Conditions of Figure III-29 except $T_I = 1$ sec.

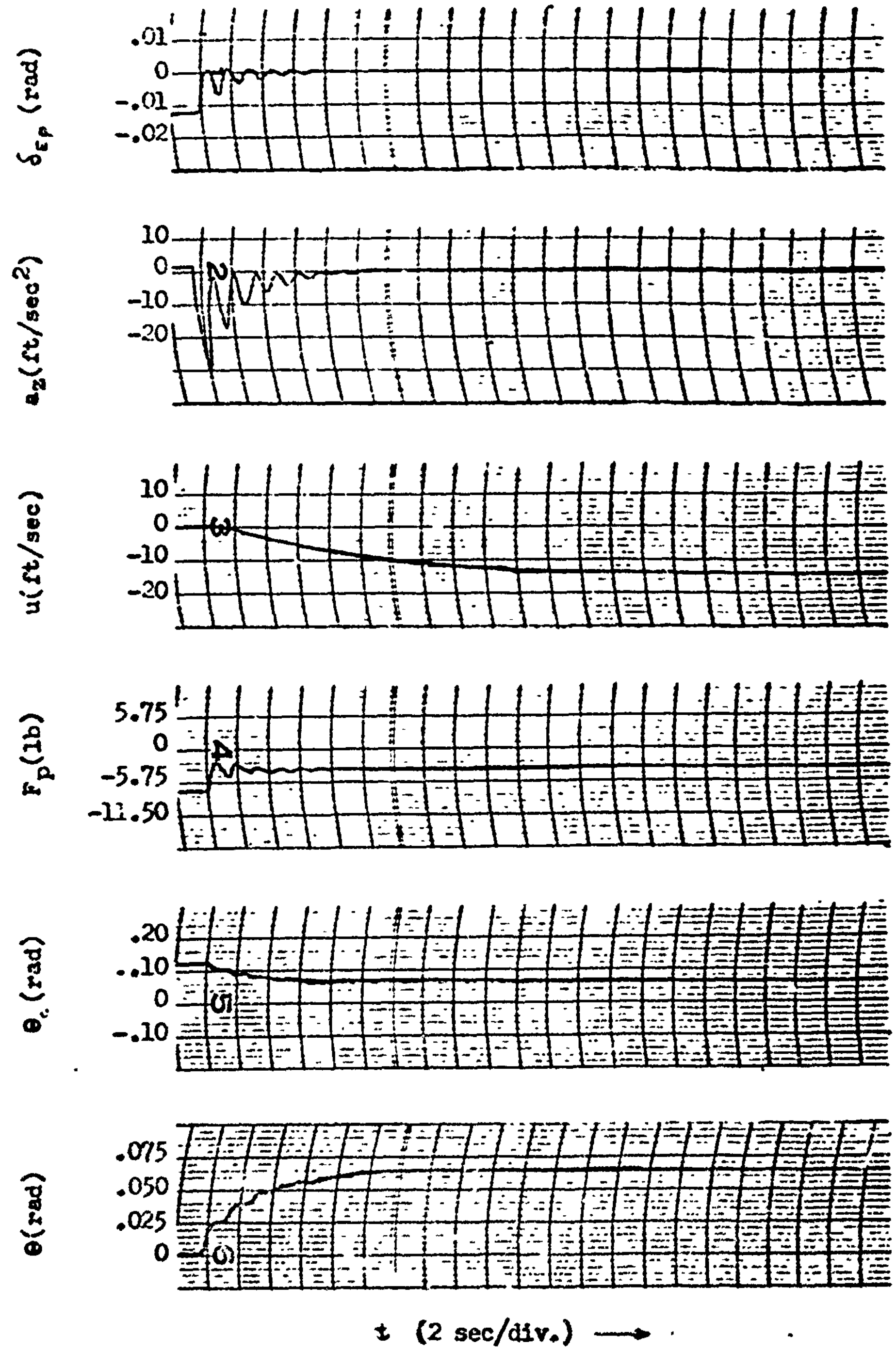


Figure III-44. Response of Pilot-Augmented Airframe System to Step θ_{ref} Command for Conditions of Figure III-29 except T_1 = 2 sec.

there is practically no change in the system response curves. It is reasonable to expect then that system augmentation can be accomplished by using an average value for T_2 with no detrimental effect on the final response when T_2 varies.

Figures III-46 through III-50 show the effect of varying the pilot's gain term K without a commensurate change in the spring constant K_{δ_e}, i.e., of varying the system gain, as indicated in (III-40a). It will be noted that as K is increased, all the quantities increase by proportionate amounts. The most significant point is that in the steady state, θ more nearly approaches θ_{ss} as K is increased, i.e., $\theta_{\epsilon_{ss}}$ decreases.* However, there is an upper limit to K, since instability will set in if K is made too large. This trend is indicated in Figure III-50, where for $K = 48$, a second hump, indicative of decreased damping, can be seen in the a_z trace.

To illustrate that augmentation can make two radically different systems behave similarly, another set of airframe conditions will be chosen. The airframe parameters are given in Table III-4. For these values, the airframe phugoid mode is of a very long period and lightly damped, while the short period mode is completely damped out in one cycle as indicated in Figure III-51. This figure should be compared with Figure III-28 to note the differences in the airframe responses.

$U = 486$	$Z_u = -.15$	$M_u = -.005$
$X_u = -.01$	$Z_w = -.53$	$M_{\dot{w}} = -.0002$
$X_w = .029$	$Z_{\dot{\theta}} = -2.93$	$M_{\dot{\theta}} = -1.03$
	$Z_{\delta_e} = 17.2$	$M_{\delta_e} = 6.5$

Table III-4. Airframe Parameters Used in Figures III-51 through III-56

* See Reference 1, p. IV-3.

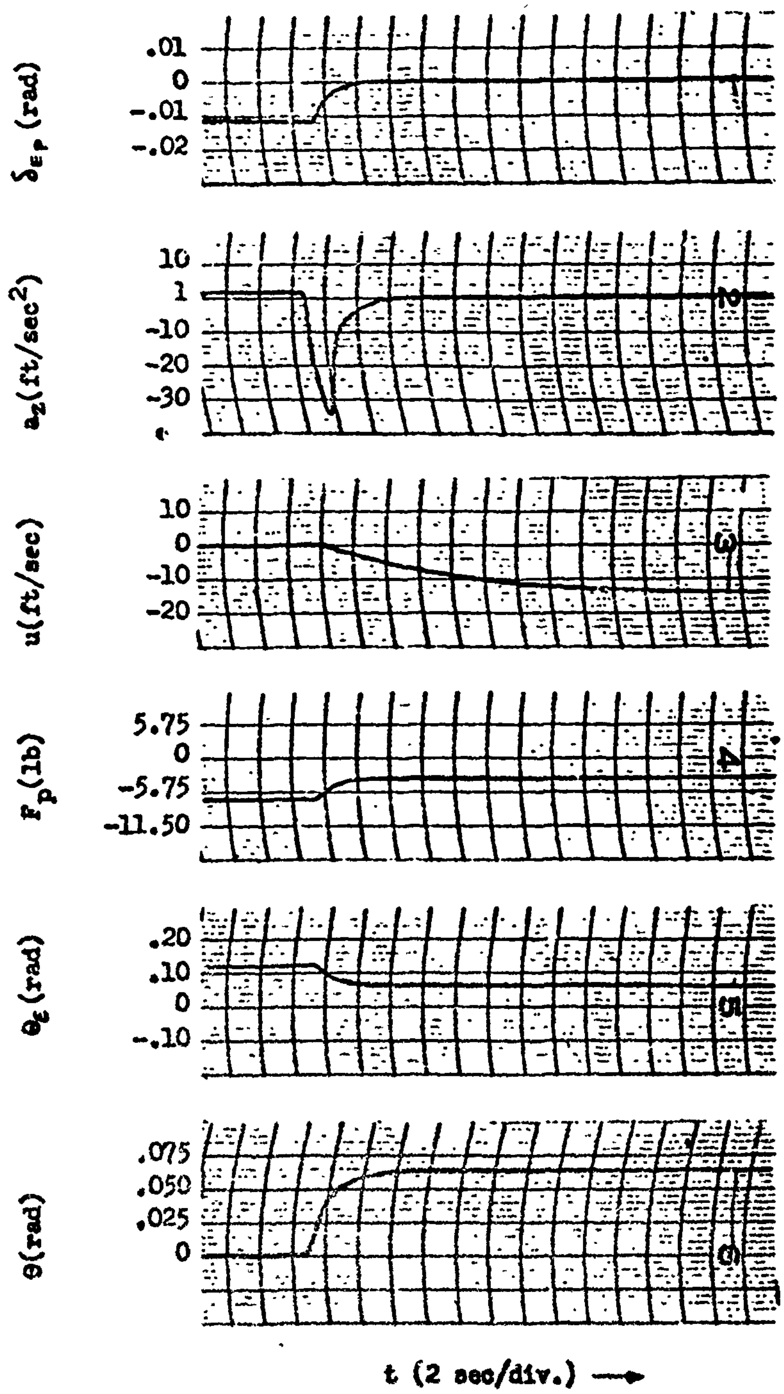


Figure III-45. Response of Pilot-Augmented Airframe System to Step θ_{ref} Command for Conditions of Figure III-29 except $T_z = 1$ sec.

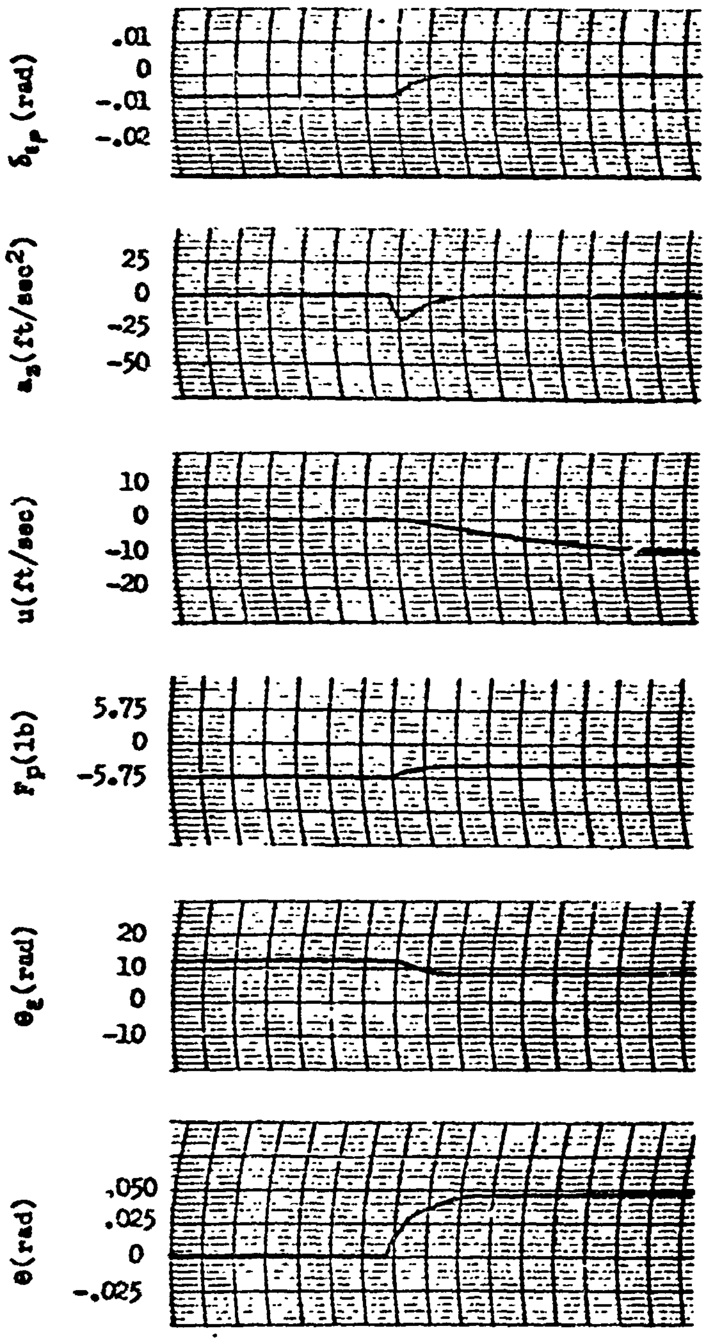


Figure III-46. Response of Pilot-Augmented Airframe System to Step θ_{ref} Command for Conditions of Figure III-29 except K = 24.

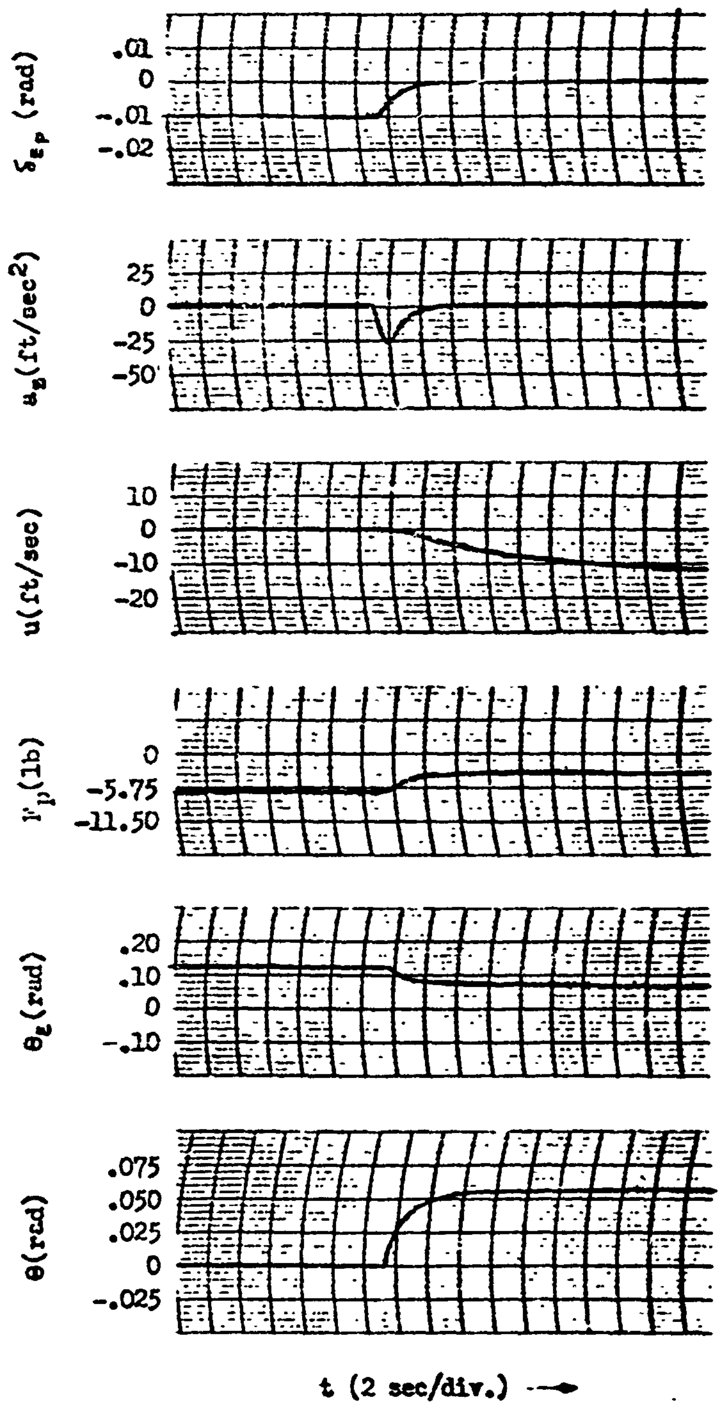
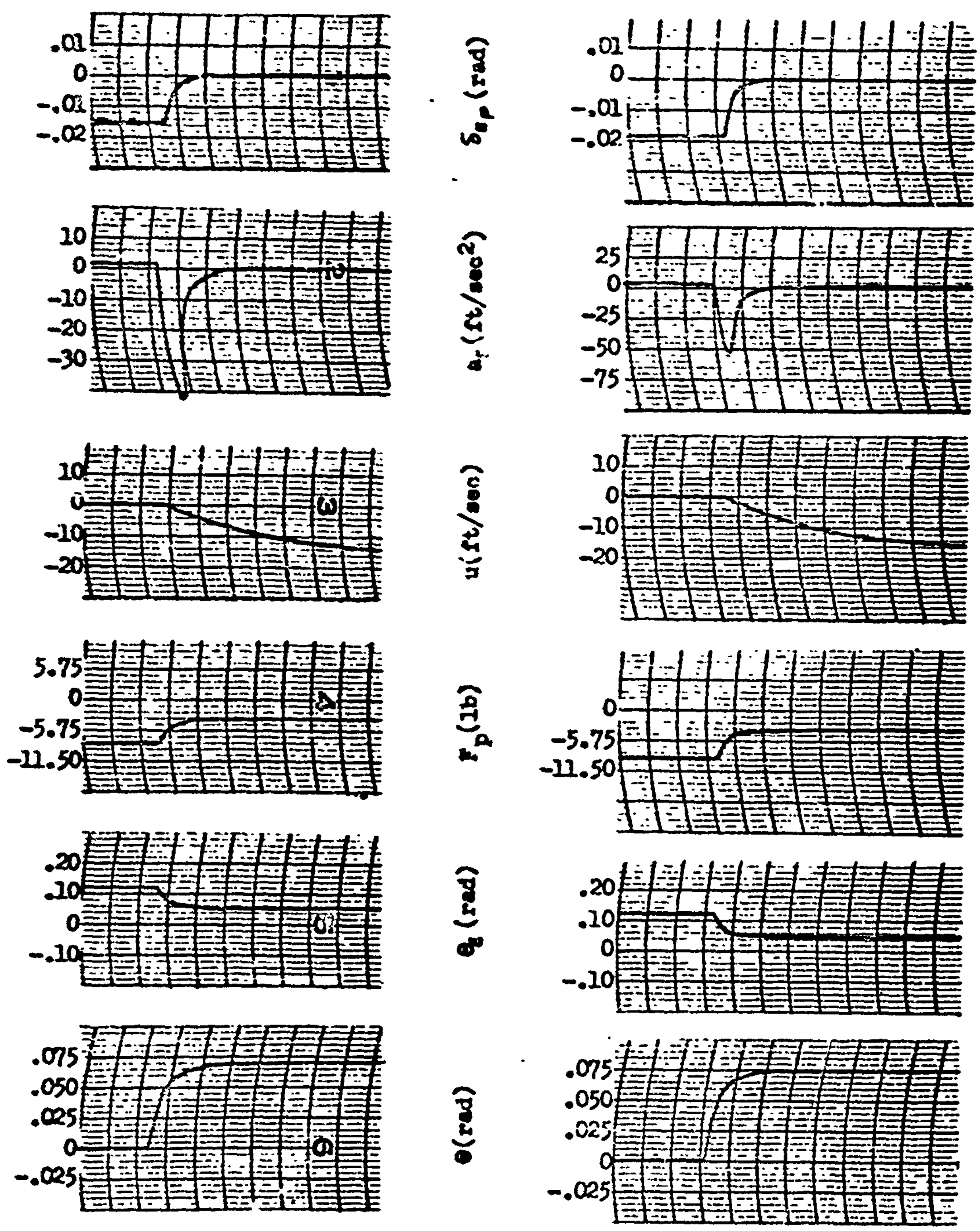


Figure III-47. Response of Pilot-Augmented Airframe System to Step θ_{ref} Command for Conditions of Figure III-29 except K = 20.



K = 36

K = 40

t (2 sec/div.) ⟶

Figure III-48

Figure III-49

Response of Pilot-Augmented Airframe ystem to Step θ_{ref} Command for Conditions of Figure III-29.

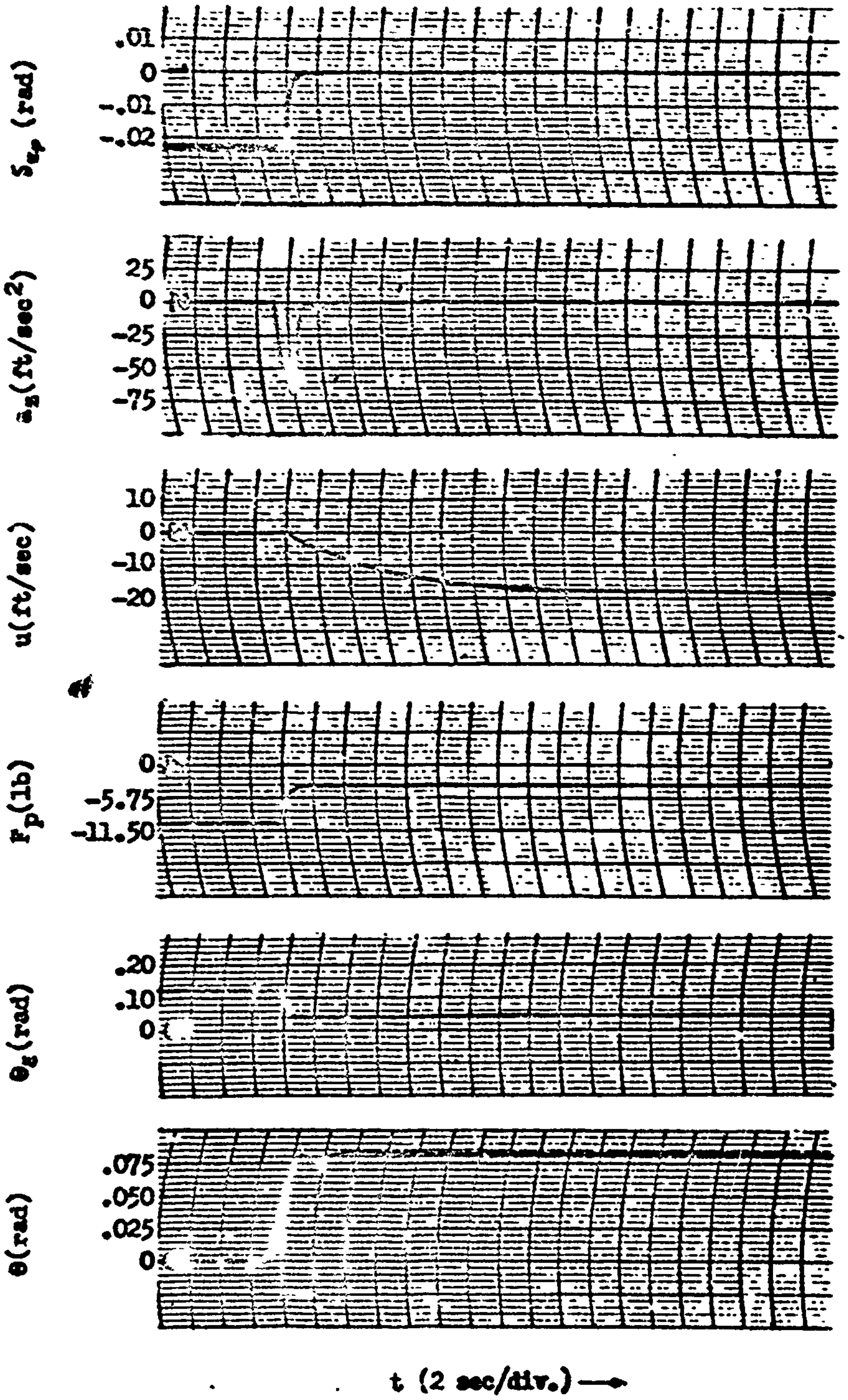


Figure III-50. Response of Pilot-Augmented Airframe to Step θ_{ref} Command for Conditions of Figure III-29 except $K = 48$.

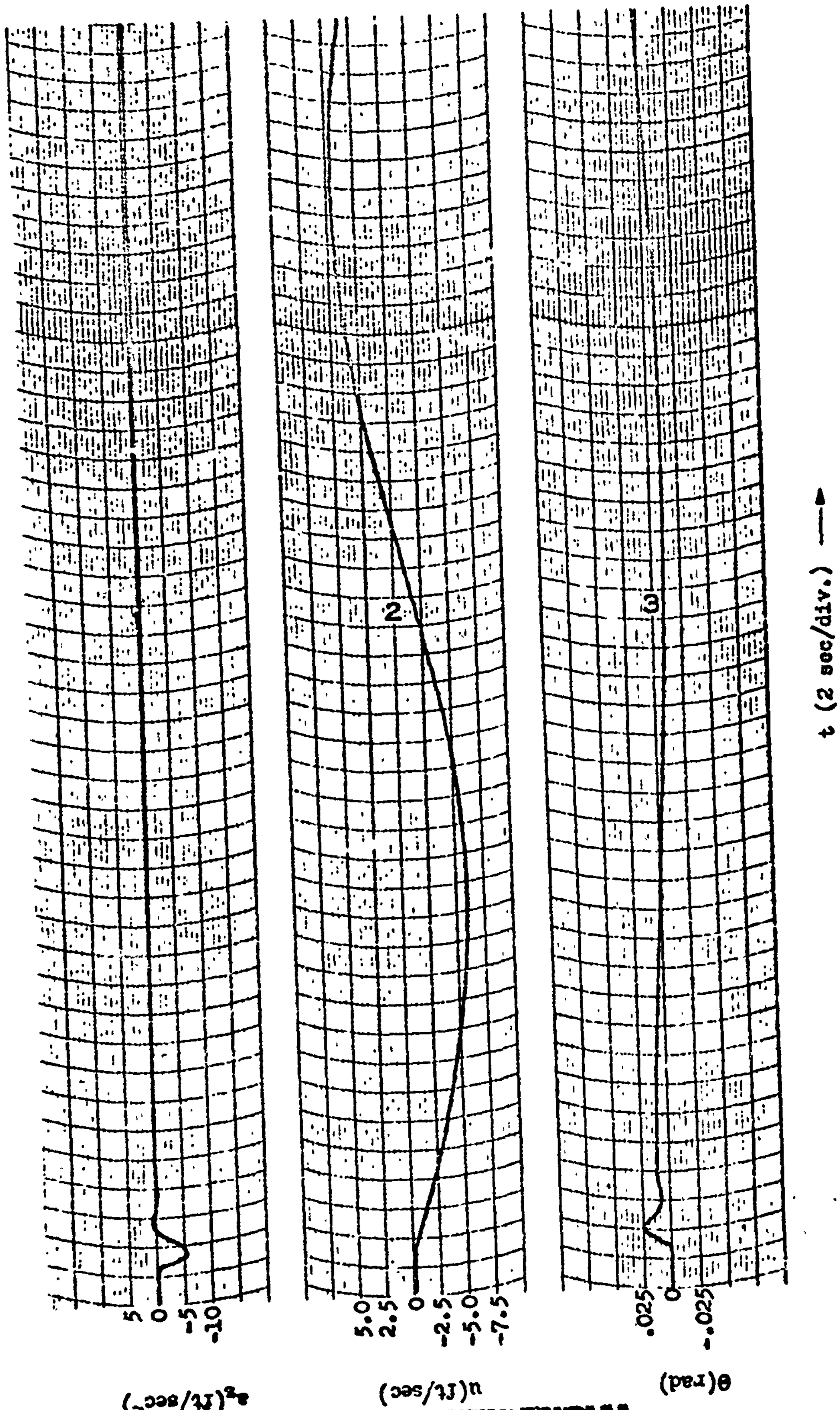


Figure III-51. Response of Unaugmented Airframe to $\ddot{\theta}$ Gust Input for Conditions of Table III-4.

Figure III-52 shows the time response of the pilot attempting to bring the system up to a new pitch attitude. The curves are very similar to the ones presented in Figure III-19. There are a few short period wiggles in the transient stage and a long period oscillation about the steady state value of Θ. This figure should be compared with Figure III-29.

Using (III-55) the stick force per g value is approximately 9.4 lb/g. To lower this value, it was found necessary to feed back some negative a_z besides using $\dot{a}_z$ and $\dot{u}$ augmentation to increase the system damping.

Figure III-53 gives the time history of the pilot-airframe system with what is considered optimum augmentation for this craft. The similarity between Figure III-53 and Figure III-34 should be noted. The augmenting values used in Figure III-53 are

$$Z_u' = -.001 \qquad K_{u_M} - \frac{K_{u_P}}{K_{\delta_s}} = -.000058 \ \frac{\text{rad}}{\text{ft/sec}}$$

$$Z_{\dot{u}} = .025 \qquad K_{\dot{u}_M} = .00145 \ \frac{\text{rad}}{\text{ft/sec}^2}$$

$$Z_{a_z} = -.015 \qquad K_{a_{z_M}} - \frac{K_{a_{z_P}}}{K_{\delta_s}} = -.00087 \ \frac{\text{rad}}{\text{ft/sec}^2}$$

$$Z_{\dot{a}_z} = .0115 \qquad K_{\dot{a}_{z_M}} = .00067 \ \frac{\text{rad}}{\text{ft/sec}^3}$$

Although this was not a tuck-under condition, it was found necessary to use some u feedback. This can be seen from Figures III-54 and III-55. In Figure III-54, with no u feedback, the θ response tends to drift back after reaching a peak. In Figure III-55, with $Z_u' = -.002$, there is an initial sharp rise in θ, after which θ very slowly increases to the steady state value. Comparison of Figures III-53, III-54, and III-55 shows that $Z_u' = -0.001$ gives the best response.

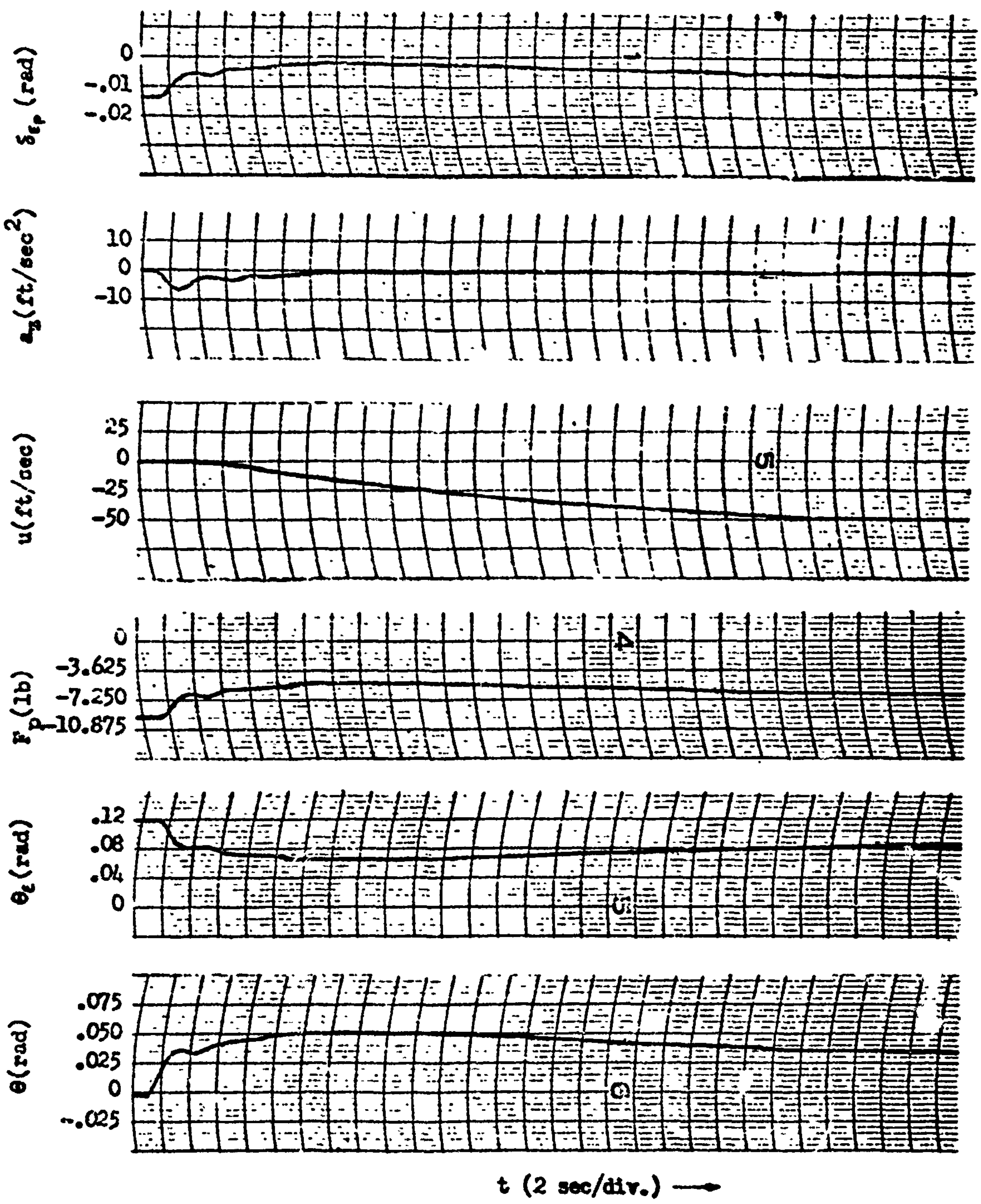


Figure III-52. Response of Pilot-Unaugmented Airframe System to Step θ_{ref} Command for Conditions of Table III-4 ($K = 32$, $\tau = 0.3$, $T_1 = 0$, $T_2 = 0.2$ sec).

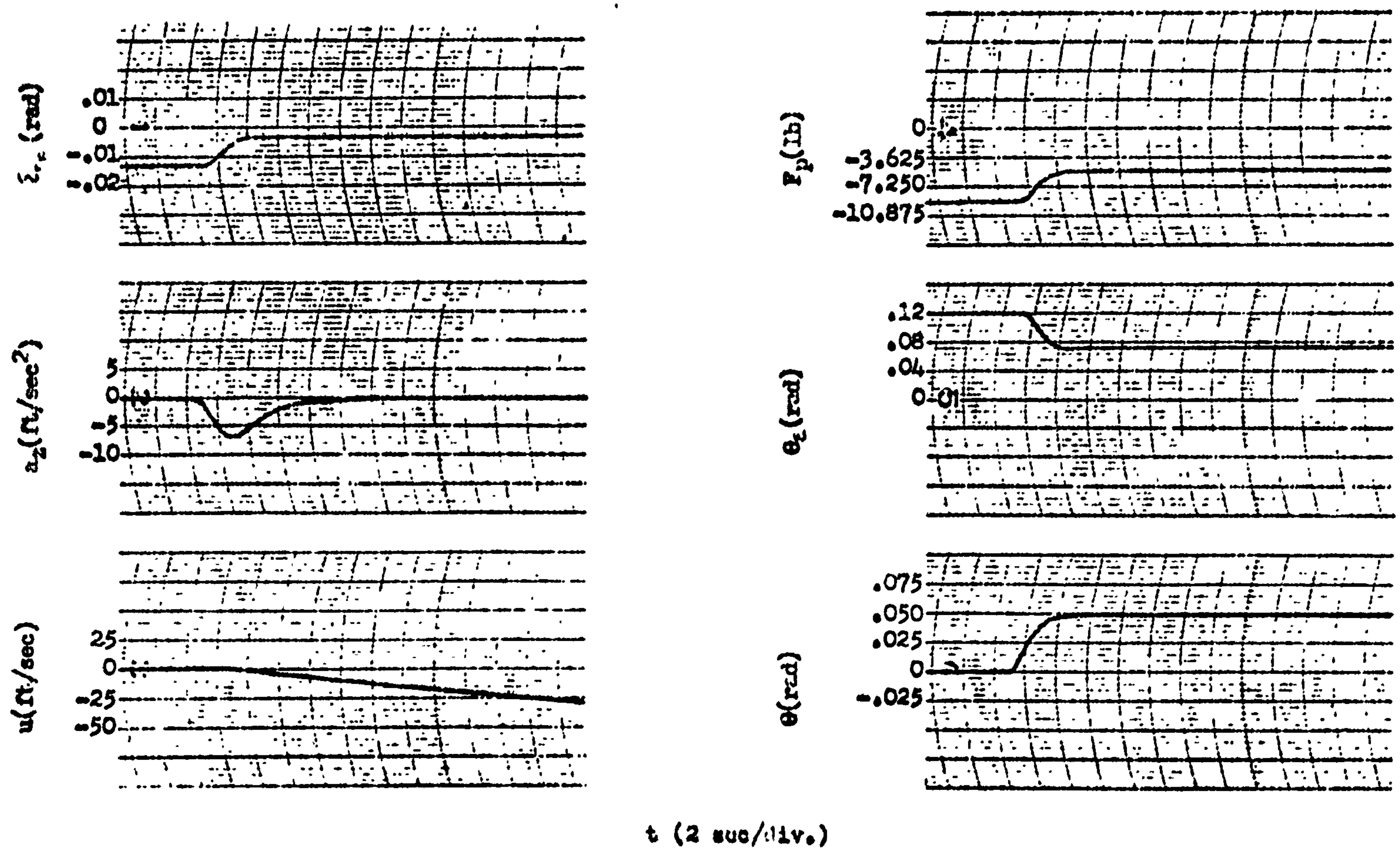


Figure III-53. Response of Pilot-Augmented Airframe System to Step θ_{ref} Command for Conditions of Figure III-52 ($Z_u' = -.001$, $Z_{\dot{\alpha}} = .025$, $Z_{a_s} = -.015$, $Z_{\dot{h}_s} = .0115$).

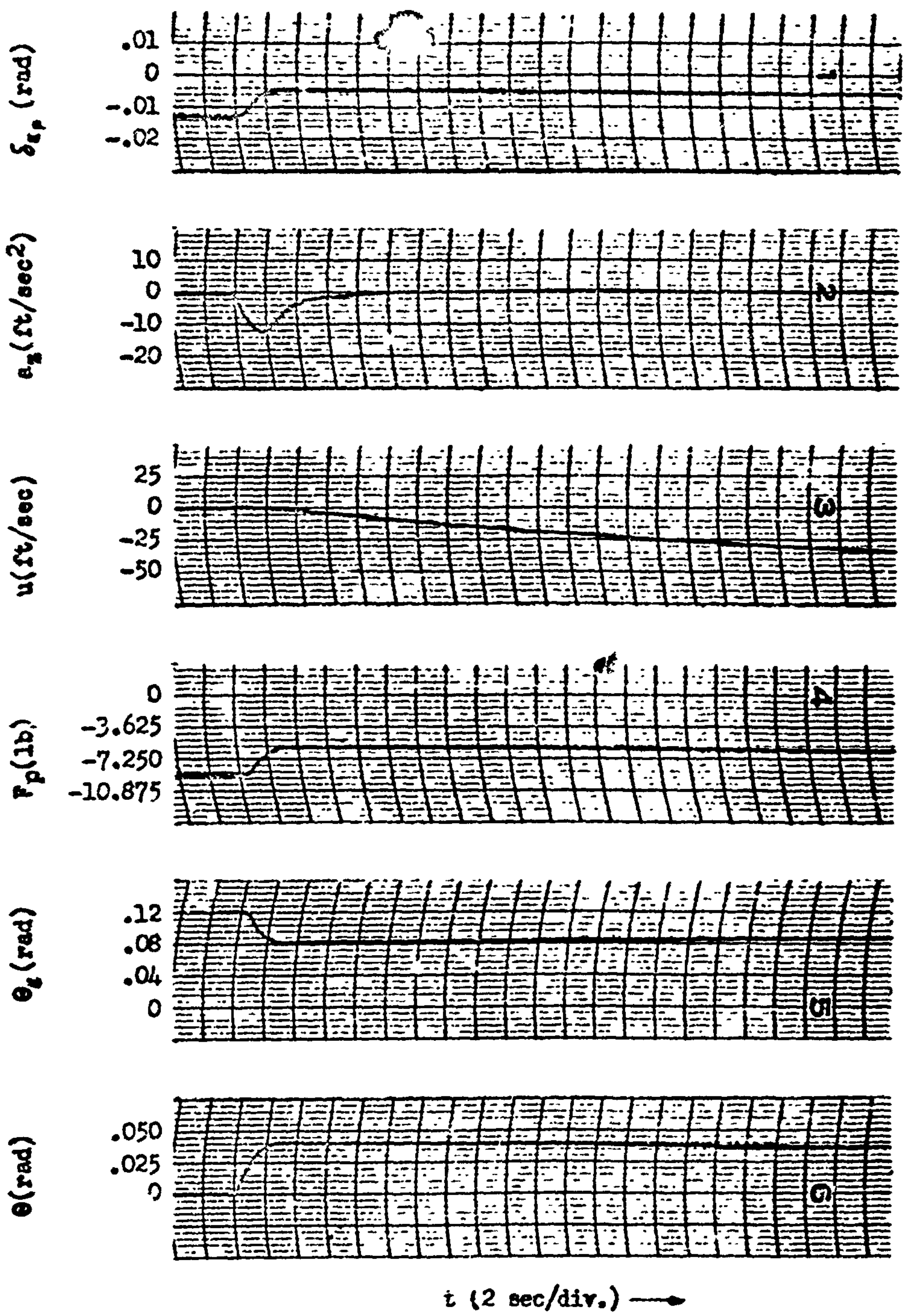


Figure III-54. Response of Pilot-Augmented Airframe System to Step θ_{ref} Command for Conditions of Figure III-53 except $Z_u' = 0$.

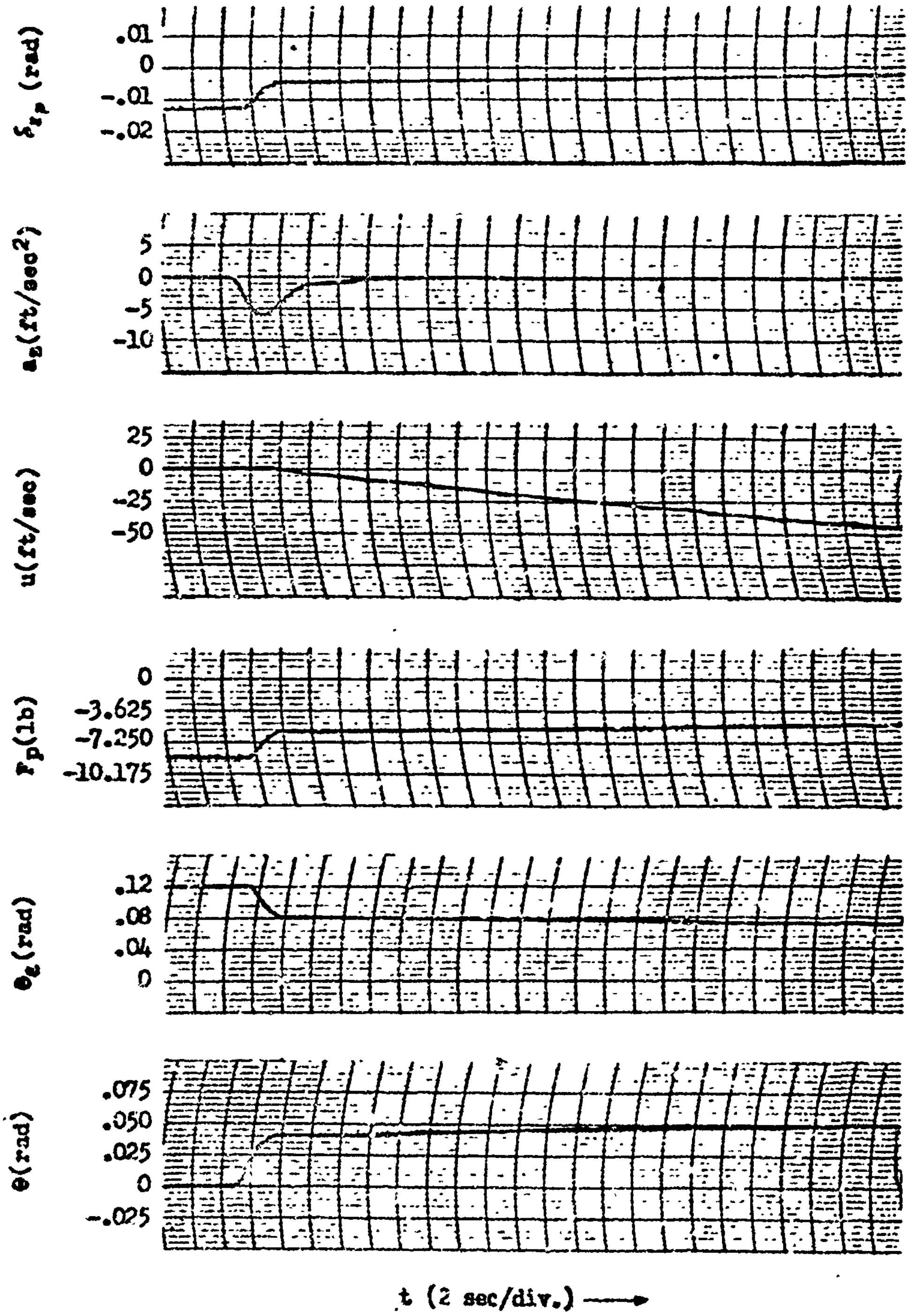


Figure III-55. Response of Pilot-Augmented Airframe System to Step θ_{ref} Command for Conditions of Figure III-53 except $Z_u' = -.002$.

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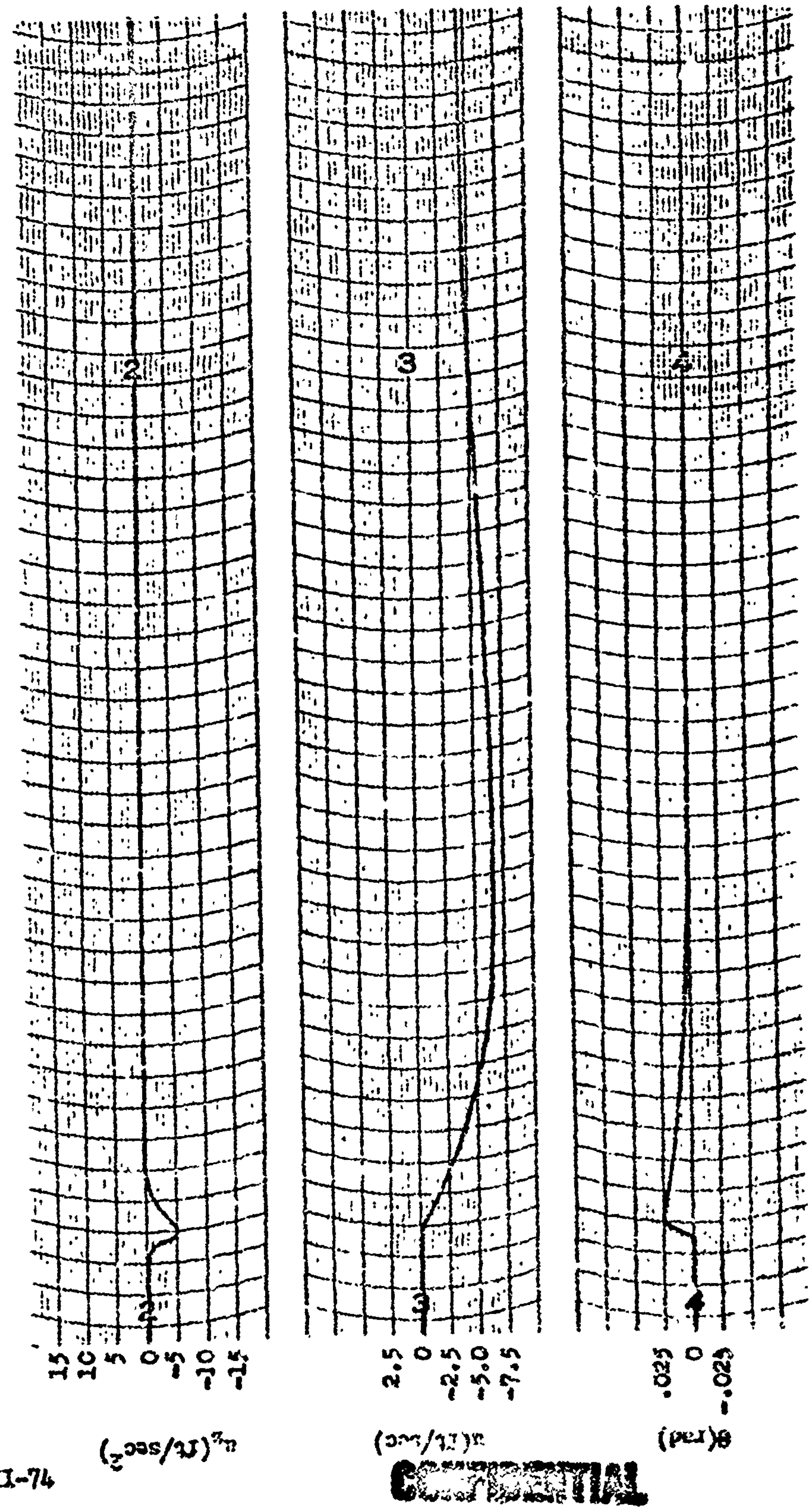

Figure III-56. Response of Augmented Airframe to 0 Gust Input

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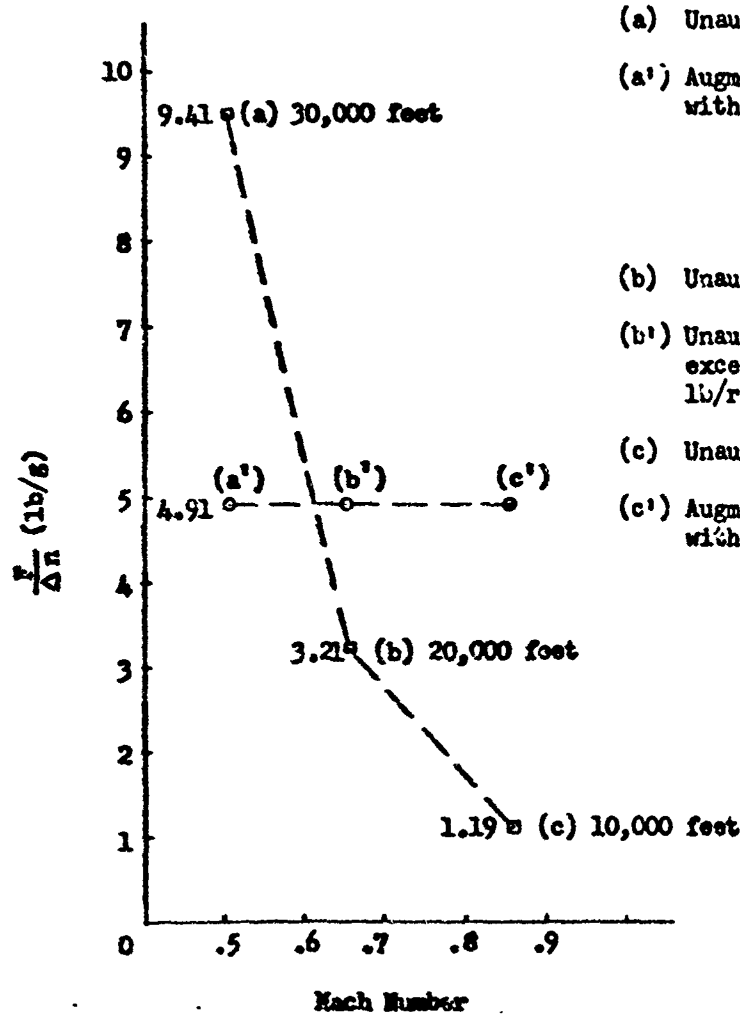
Figure III-56 shows the equivalent airframe without pilot control when disturbed by a vertical gust of wind. Note that the augmentation has stabilized the system to a great extent, eliminating both phugoid and short period oscillations completely. Compare Figure III-56 with Figure III-51.



(a) Unaugmented airframe of Table III-4.

(a') Augmented airframe of Table III-4 with:

$K_{\delta_s} = 160$ lb/rad $= 2.8$ lb/deg.

$K_{a_z} Z_{\delta_s} = -.015$

$K_{a_z} = -.00087 \ \frac{\text{rad}}{\text{ft/sec}^2}$

(b) Unaugmented airframe of Table III-2.

(b') Unaugmented airframe of Table III-2 except K_{δ_s} increased to 245 lb/rad $\cong 4.3$ lb/deg.

(c) Unaugmented airframe of Table III-3.

(c') Augmented airframe of Table III-3 with:

$K_{\delta_s} = 470.4$ lb/rad $= 8.2$ lb/deg.

$K_{a_z} = .0000932 \ \frac{\text{rad}}{\text{ft/sec}^2}$

Figure III-57. Effects of Augmentation on Stick Force per g

It will be of interest at this point to investigate the effects of augmentation on the stick force per g values for the three sets of airframe parameters considered. Using (III-55), the points in Figure III-57 have been calculated for both augmented and unaugmented airframes. Note that by using both a_z augmentation and variation of K_{δ_z}, the stick force per g characteristic can be made the same for all three conditions. The level of the constant stick force per g curve can be shifted up or down by merely varying K_{δ_z}.

In conclusion, it can be said that by the method of equivalent stability derivatives and analog computer simulation, a complete preliminary study can be made to determine what sort of equalization is required to optimize the pilot-airframe system of Figure III-2. From this preliminary study, the variation of $K_{u_{sw}}$, K_{u_p}, $K_{\dot{u}_{sw}}$, $K_{a_{z_{sw}}}$, $K_{\dot{a}_{z_{sw}}}$, $K_{a_{z_p}}$, and K_{δ_z} with Mach number and altitude can be estimated. The problem of system mechanization then remains.

SECTION 4 - SYSTEM MECHANIZATION

From all indications, the gains of the feedback quantities will have to be programmed with Mach number and altitude and/or dynamic pressure in order to establish optimum response characteristics for different flight conditions. This type of mechanization has not proved to be of any difficulty in the past and should not present any problems now.

It remains then to select the sensors and actuators to be used in the equalization. Consider first the sensors. For the equalization chosen, there need be only a normal accelerometer and some sort of forward velocity pickup to give output voltages proportional to a_z and u. These voltages, when sent through variable rate circuits, will give voltages proportional to $\dot{a}_z$ and $\dot{u}$ as well as a_z and u as shown in Figure III-58.

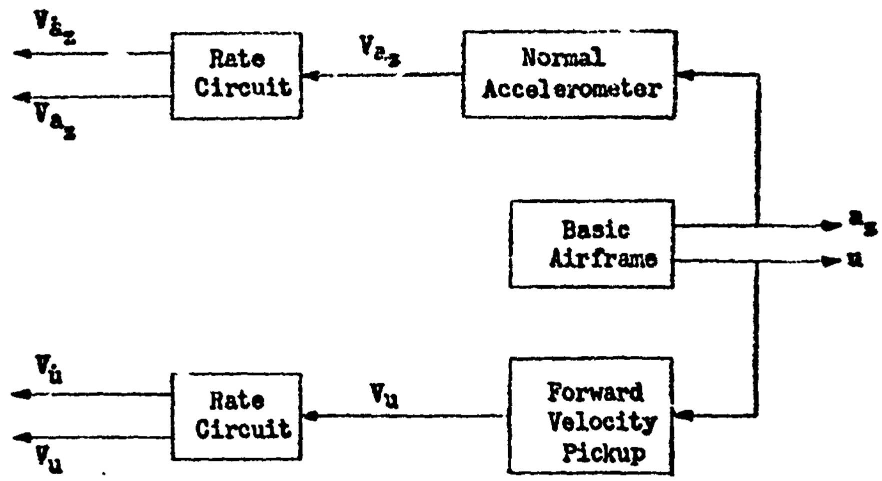


Figure III-58. Block Diagram Showing the Generation of Voltages Proportionate to u , $\dot{a}$, a_z , and $\dot{a}$.

Consider now V_u and V_{a_z} . They can be used as activating signals for a force producing device in place of the commonly accepted bobweights, centering springs, etc., if it is so desired. The force producing device could be a hydraulic cylinder with an electrically operated valve. The piston rod attached to the control stick can be made to exert a feel force proportional to V_{a_z} and V_u as indicated in Figure III-59.

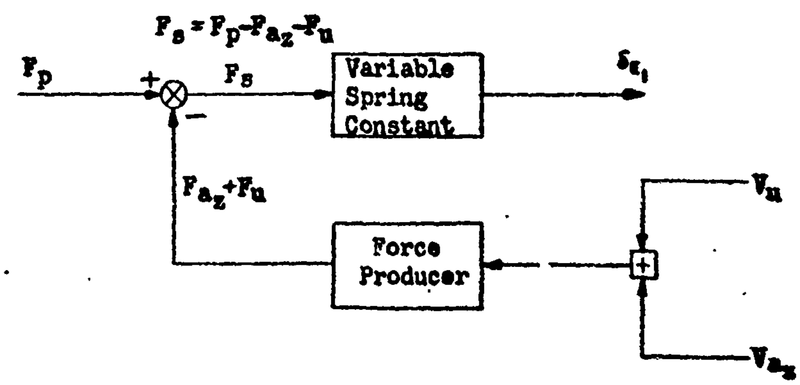


Figure III-59. Block Diagram Showing Mechanization of u and a_z Force Feedbacks to the Stick

The variations in K_{δ_s}, the control system spring constant, can be accomplished by using a bellows arrangement which effectively increases the spring constant as dynamic pressure is increased. The values for K_{δ_s} given in Figure III-57 are satisfied by

$$K_{\delta_s} = (2 + 0.007\ q_c)\ \text{lb/deg.} \quad \text{(III-56)}$$

where

q_c = dynamic pressure (lb/ft^2).

The motion stability augmenter is arranged in a series installation with the other components of the control system. That is, any deflections of the elevator caused by the motion stability augmenter are not reflected back through the spring. This is desirable since the main purpose of the motion stability augmenter is to damp out unwanted airframe motions. The pilot should not be annoyed by unexpected stick deflections whenever the elevator is moved by the stability augmenter.

The activating signals for the motion augmenter come from V_u, $V_{\dot{u}}$, V_{δ_s}, and $V_{\dot{\delta}_s}$. These signals control an electrically operated hydraulic cylinder whose output motion deflects the elevator. The complete pilot-equivalent airframe system is shown in Figure III-60.

An important point to be considered is that the sensors and actuators are not perfect; i.e., they contain inherent lags, thresholds, and other nonlinearities. The final system configuration must be based on a study which includes all these effects plus any additional equalization that is required to compensate for the component lags.

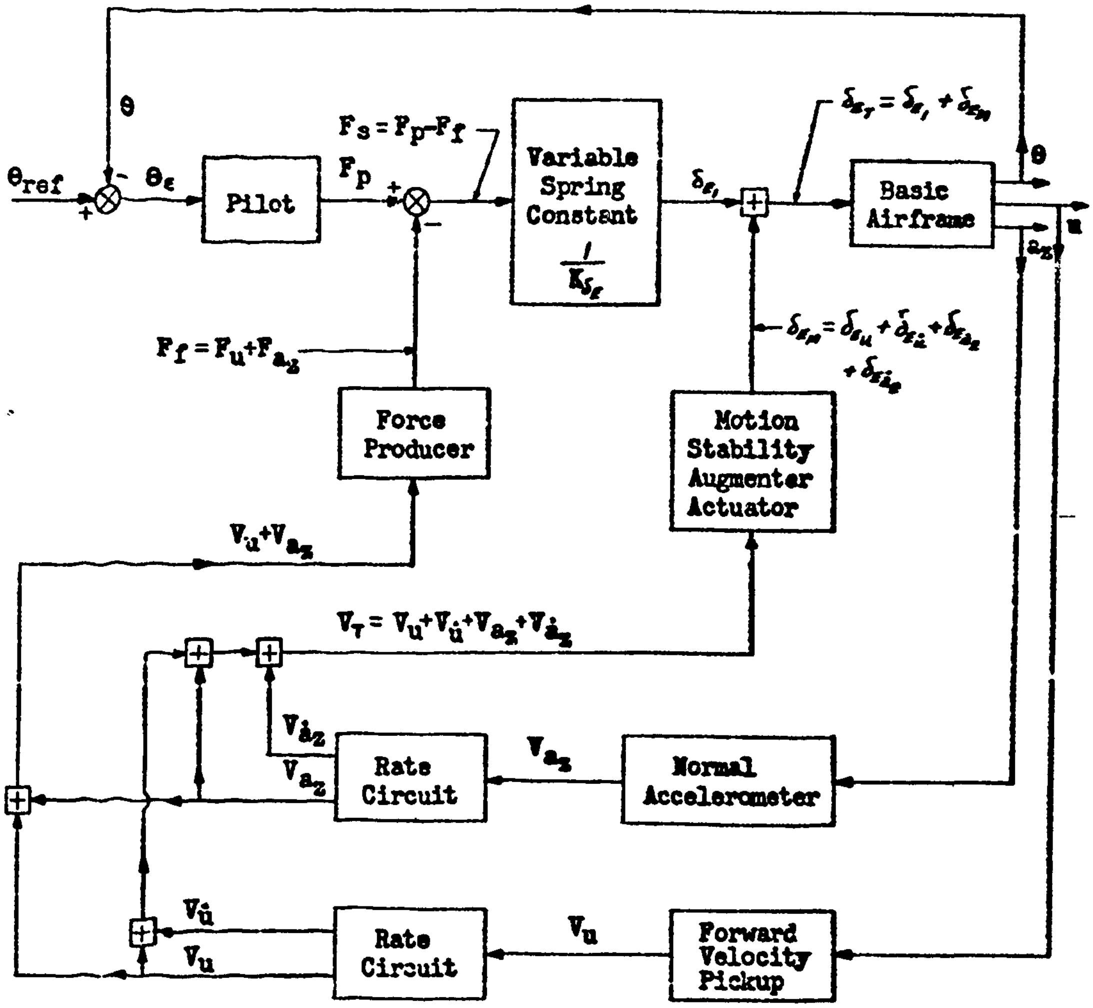


Figure III-60. Block Diagram of Pilot-Equivalent Airframe System

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1. Methods of Analysis and Synthesis of Piloted Aircraft Flight Control Systems, BuAer Report No. AE-61-4 I, Northrop Aircraft, Inc., Hawthorne, California, March 1952.

2. Mead, R. J., "Simulation of the Human Pilot for Straight Flight with One Degree of Freedom," Goodyear Report No. AP-23633, Goodyear Aircraft Corp., Akron, Ohio, November 6, 1951. (Confidential)

3. Final Report: Human Dynamics Study, BuAer Contract No. NOa(s)-51-083-c, Goodyear Report No. GER-4750, Goodyear Aircraft Corp., Akron, Ohio, April 8, 1952. (Confidential)

4. Dynamics of the Airframe, BuAer Report No. AE-61-4 II, Northrop Aircraft, Inc., Hawthorne, California, Septermber 1952.

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CHAPTER IV

DESIGN CRITERIA

SECTION 1 - INTRODUCTION

At the present time, there are several sets of requirements for the flying qualities of piloted aircraft. The purpose of this chapter is to integrate and codify these requirements.

Section 2 presents a general discussion of the requirements, and Sections 3 and 4 give more detailed discussions devoted respectively to the longitudinal and lateral-directional requirements. Section 5 includes some recommendations and suggestions for further study.

SECTION 2 - GENERAL DISCUSSION

It was pointed out in Chapter III that the present specifications for flying qualities of piloted aircraft have been based on a series of flight test investigations and on the resulting pilots' opinions. On the basis of these studies, and bearing in mind such factors as

1. Pilot safety and comfort,
2. Pilot capabilities,
3. Airframe safety,
4. Maneuverability, and
5. Ease of maintaining a given attitude,

desirable stability and control characteristics can be formulated. These characteristics are codified in Tables IV-1 and IV-2 and are more fully discussed in Sections 3 and 4. Tables are appended at the end of this chapter.

SECTION 3 - LONGITUDINAL REQUIREMENTS

The pilot-equivalent airframe system is shown as a block diagram in Figure IV-2.

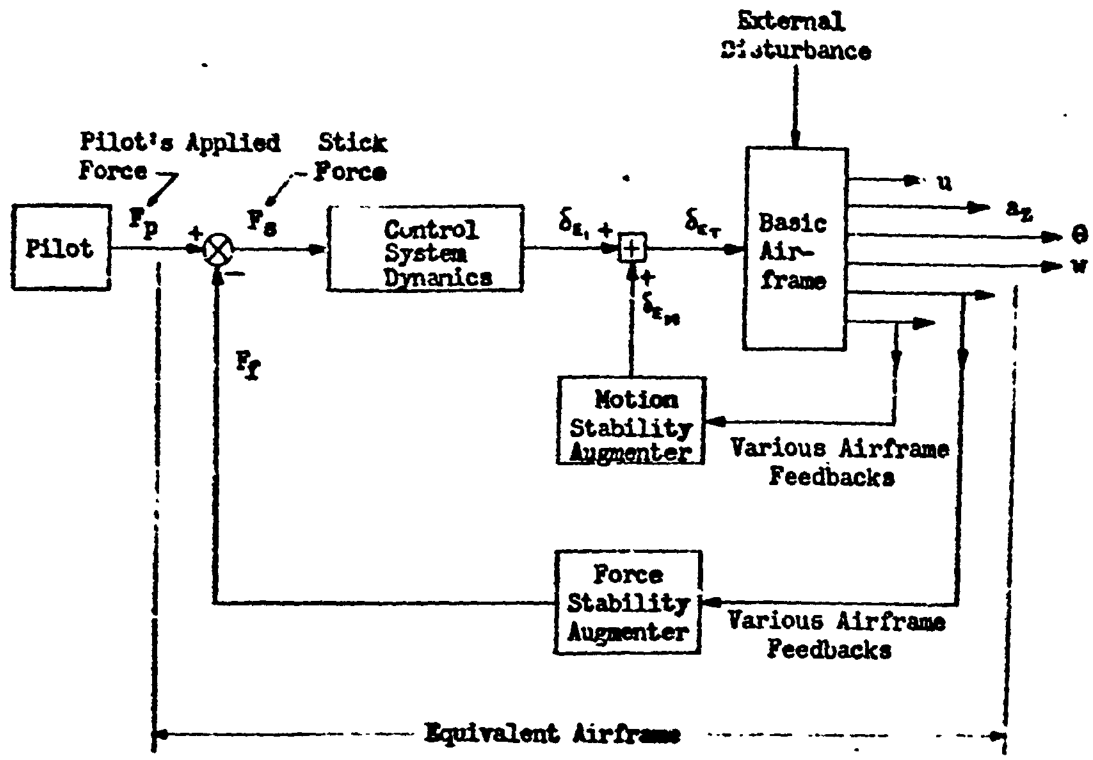

Figure IV-2. Block Diagram of Pilot-Equivalent Airframe (Longitudinal) System

The longitudinal requirements for stability and control given in Table IV-1 should be examined with this figure in mind.

(a) DYNAMIC STABILITY

Consider first the dynamic stability specification. In essence, this requirement states that the oscillations of u, a_z, Θ, and w following a disturbance should die out in a reasonably short time. The disturbance may be external, as is the case in flight through rough air, or it may be internal, as is the case when the pilot applies a control force.

As indicated in Table IV-1, all the factors given in Section 2 must be considered in the specification for short period dynamic oscillations. From

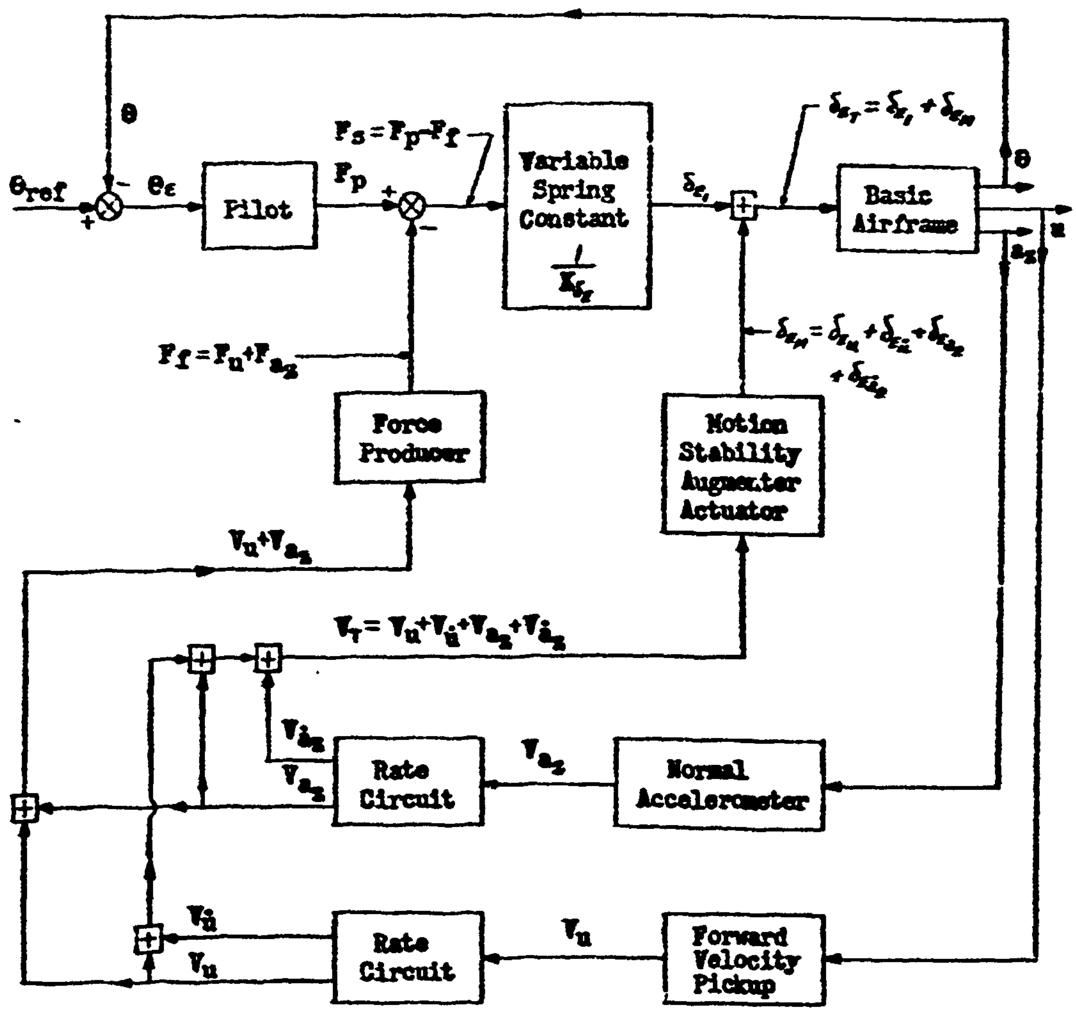

Figure III-60. Block Diagram of Pilot-Equivalent Airframe System

BIBLIOGRAPHY

1. Methods of Analysis and Synthesis of Piloted Aircraft Flight Control Systems, BuAer Report No. AE-61-4 I, Northrop Aircraft, Inc., Hawthorne, California, March 1952.

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3. Final Report: Human Dynamics Study, BuAer Contract No. NOa(s)-51-083-c, Goodyear Report No. GER-4750, Goodyear Aircraft Corp., Akron, Ohio, April 8, 1952. (Confidential)

4. Dynamics of the Airframe, BuAer Report No. AE-61-4 II, Northrop Aircraft, Inc., Hawthorne, California, September 1952.

DESIGN CRITERIA

SECTION 1 - INTRODUCTION

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It was pointed out in Chapter III that the present specifications for flying qualities of piloted aircraft have been based on a series of flight test investigations and on the resulting pilots' opinions. On the basis of these studies, and bearing in mind such factors as

1. Pilot safety and comfort,
2. Pilot capabilities,
3. Airframe safety,
4. Maneuverability, and
5. Ease of maintaining a given attitude,

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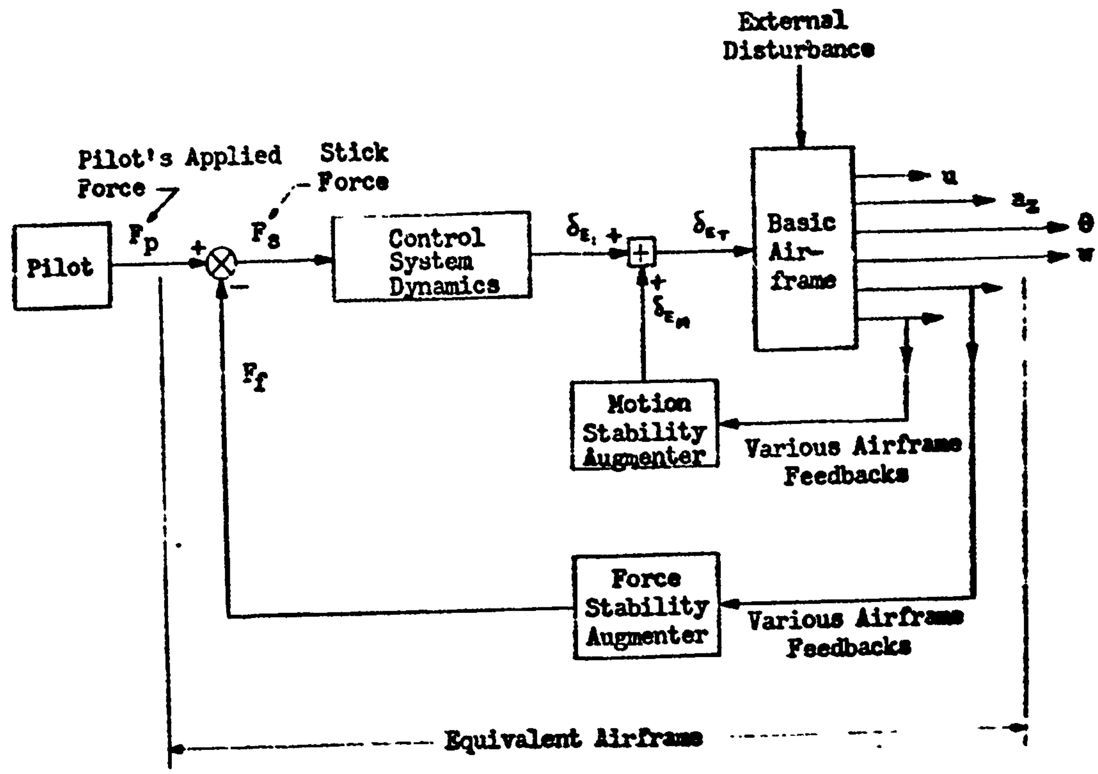

Figure IV-2. Block Diagram of Pilot-Equivalent Airframe (Longitudinal) System

The longitudinal requirements for stability and control given in Table IV-1 should be examined with this figure in mind.

(a) DYNAMIC STABILITY

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As indicated in Table IV-1, all the factors given in Section 2 must be considered in the specification for short period dynamic oscillations. From

the standpoint of pilot safety and comfort, it is obvious that any sustained oscillations of normal acceleration acting on the pilot can become quite uncomfortable, possibly leading to vertigo and loss of control. For this reason, the short period should be heavily damped.

Another reason for requiring heavy damping for the short period is the limit of the pilot's capabilities. When the frequency of oscillation reaches approximately one cycle per second, the human pilot is not capable of controlling the oscillation unless the oscillatory mode is well damped to begin with. This fact was illustrated in Figures III-29, III-30, III-31, and III-32.

The need for heavy damping of the short period normal acceleration oscillation is most easily appreciated from the standpoint of airframe safety. Consider the case where an airframe which is unstable in the short period mode is excited by a gust of wind. The disastrous effect of a diverging short period oscillation is apparent.

With regard to maneuverability, well-damped airframe oscillatory modes lead naturally to minimization of hunting when new steady state attitudes are sought.

At the moment, the degree of damping required for the phugoid mode is conjectural. As long as the phugoid mode is not diverging or is not left uncontrolled, the pilot is not bothered nor is the performance of any specific mission hampered. However, if the lightly damped phugoid oscillation is not controlled, the pilot is apt to become airsick. To prevent this, the pilot can usually damp out the phugoid oscillation by cockpit control movements.

However, if the pilot is to control the phugoid when the oscillations begin, he must consciously set out to do so. Unlike the short period mode, there are no noticeable acceleration forces acting on the pilot's body in the phugoid mode. The only indication which the pilot receives of an oscillatory phugoid mode is from the cockpit instruments, e.g., the airspeed indicator.

Therefore, in the event that future investigations show that the phugoid mode must be damped, the equivalent airframe should be designed so as to have the desired phugoid characteristics.

(b) STATIC STABILITY

In terms of the equivalent airframe block diagram, the static stability requirement states that a pull force, F_p , exerted by the pilot, which results in an up-elevator movement, will cause a negative u ($U = U_0 + u$; U = total forward speed, U_0 = trim forward speed, u = incremental forward speed). Furthermore, when the pilot has his hands off the stick, i.e., when $F_p = 0$, any perturbation, u , from trim forward speed due to any sort of external disturbance should reduce to zero.

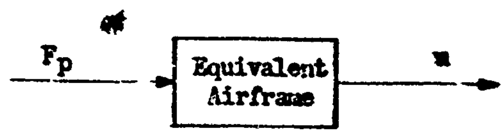

Figure IV-3. Block Diagram Used to Illustrate Static Stability Requirement

The first requirement simplifies the control movements required to initiate a change in trim speed. This follows from the fact that a change in level flight trim speed is always accompanied by a change in angle of attack. The direction of control displacement required to trim at the new angle of attack corresponds to that needed to start a rotation in pitch.

The second static stability requirement facilitates the task of maintaining a steady attitude by eliminating the need for constant monitoring of the cockpit controls.

(c) ELEVATOR CONTROL EFFECTIVENESS

The requirement that the elevator control be powerful enough to develop

maximum lift coefficient or design load factor insures that the airplane can perform up to its aerodynamic and/or structural design limits.

The landing requirement is possibly the most critical imposed on the elevator. When the airplane is close to the ground, more up-elevator is required to trim at a given attitude because of the reduction of downwash on the tail from the wing. The greatest up-elevator is required at the most forward center of gravity position at near stall conditions. If the elevator control is sufficient to meet this requirement, it will most likely satisfy the take-off requirements which specify that the elevator must have sufficient control to maintain the plane in the proper take-off attitude during the ground run.

(d) ELEVATOR CONTROL FORCES

The elevator control forces are specified to insure that the forces required of the pilot are at all times within the limits of his capabilities. Furthermore, in certain critical cases, i.e., landing and take-off, the forces should be such that one-handed flying is possible. However, if the forces required are made too low, the structural safety of the airplane will be endangered. This is reflected in the stick force per g requirement.

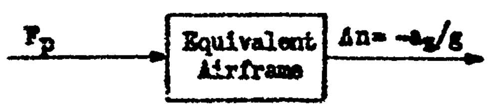

Figure IV-4. Block Diagram Used to Illust .te Stick Force per g Requirement

The force requirements state that an increase in pull force, F_p in Figure IV-4, should produce an increase in normal acceleration, Δn , and that the ratio of F_p to Δn should be greater than 3 lb /g. Assume for the moment that $F_p/\Delta n$ is 1 lb/g. Then if the pilot exerts a force of 10 pounds on the cockpit

control, Δn will be 10 g's. For an airplane which has a limit load factor of, say, 8 g's, a 10 g change in normal acceleration can lead to structural failure. For this reason, the $F_p/\Delta n$ ratio has a minimum limit specified.

The above force requirements are for steady turns and pull-ups in which the forces are approximately proportional to the changes in normal acceleration. However, in sudden pull-up maneuvers, the change in normal acceleration depends also on the elevator (and consequently on the applied force) rate of movement and exhibits a large peaking effect, as shown in Figure IV-5.

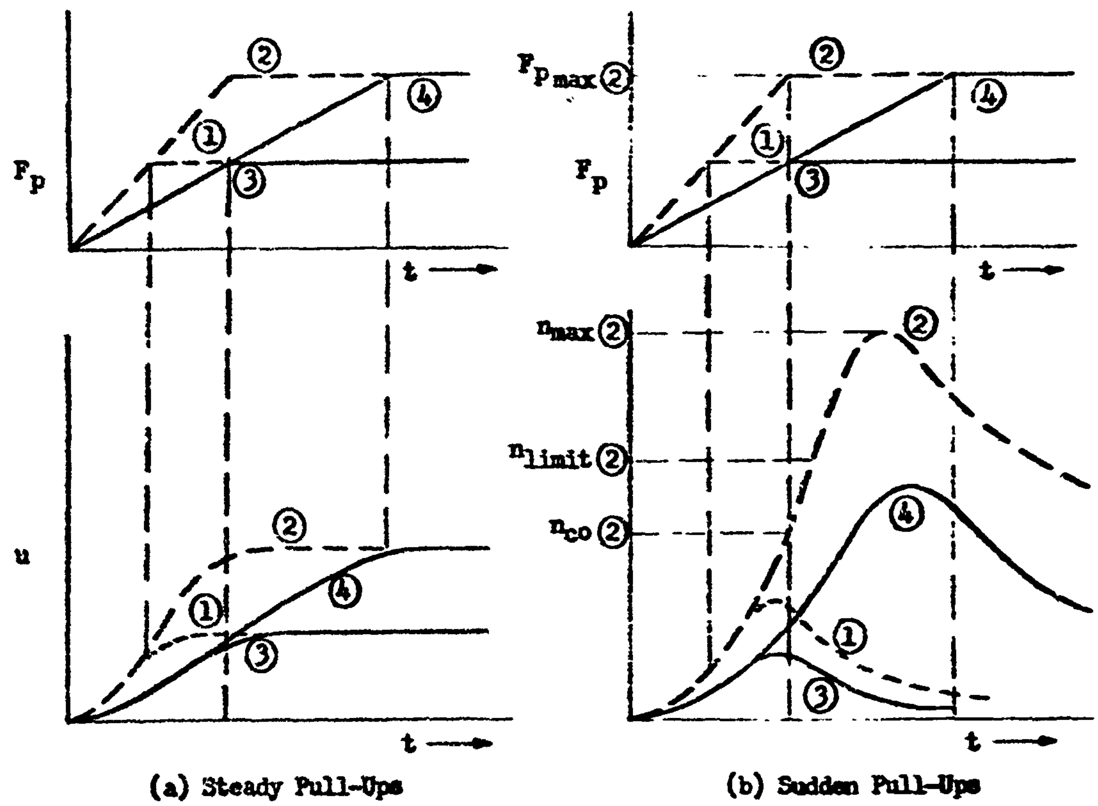

Figure IV-5. Force-Normal Acceleration Relationships

Examination of curve ② in Figure IV-5 (b) reveals several interesting points. First of all, the normal acceleration, n_{co} , at the time the elevator rate is reduced to zero gives no indication of the normal acceleration that will

eventually be experienced. Second, if the elevator rate is high enough, the maximum normal acceleration may occur after the elevator rate is zero. Last, and most important, n_{max} may exceed the limit load factor although n_{co} is well below n_{LIMIT}. This last point makes it imperative from the standpoint of airframe safety that in any sudden pull-up maneuver, $F_{p_{max}}/n_{max}$ be equal to or greater than $F/\Delta n$ for steady pull-ups.

The landing and take-off force requirements allow for one-handed flying, thus leaving the other hand free to perform other tasks. The force limits are left at reasonably high values to prevent inadvertent stalls or near stalls.

(e) LONGITUDINAL TRIM

The trim change requirement states that the equivalent airframe output quantities indicated in Figure IV-6 should be as small as possible for changes in throttle, gear, or flap setting. This requirement minimizes the effort required of the pilot to maintain a trimmed attitude during landing approaches. If any trim changes do take place, the force, F_p, required to maintain trim should not be excessive.

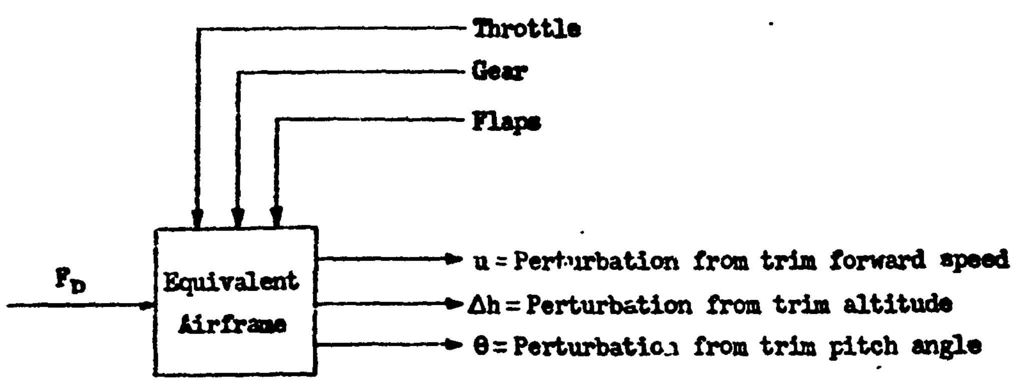

Figure IV-6. Block Diagram Used to Illustrate Longitudinal Trim Change Effects

The trim requirement for sideslips facilitates the task of maintaining trim. When a rudder deflection is initiated in a wings level flight attitude, a steady sideslip results. This in turn creates an increased drag profile and a reduction in airspeed. Consequently, the lift force decreases and unless the trim change due to sideslip is counteracted, a steady loss of altitude will result.

To cancel the loss in lift, a pull force should be required. A pull force gives more up-elevator and an increased angle of attack which leads to an increase in lift. The force required to maintain altitude in steady sideslips should be low enough so that it will not fatigue the pilot when applied for any appreciable length of time.

The steady state error in forward velocity and flight path angle specified in the trim requirements defines the amount of apparent friction in the elevator control circuit. The limitations tend to insure maintenance of trim speed and attitude.

(f) LONGITUDINAL TRIMMING DEVICES

The longitudinal trimming devices are used for the express purpose of reducing the elevator control forces to zero to relieve the pilot of continuous attention to the cockpit controls while maintaining a constant flight attitude. They should be irreversible and should hold a given setting indefinitely or until changed manually.

SECTION 4 – LATERAL-DIRECTIONAL REQUIREMENTS

The pilot-airframe block diagram for the lateral-directional case, Figure IV-7, is similar to the longitudinal block diagram (Figure IV-2).

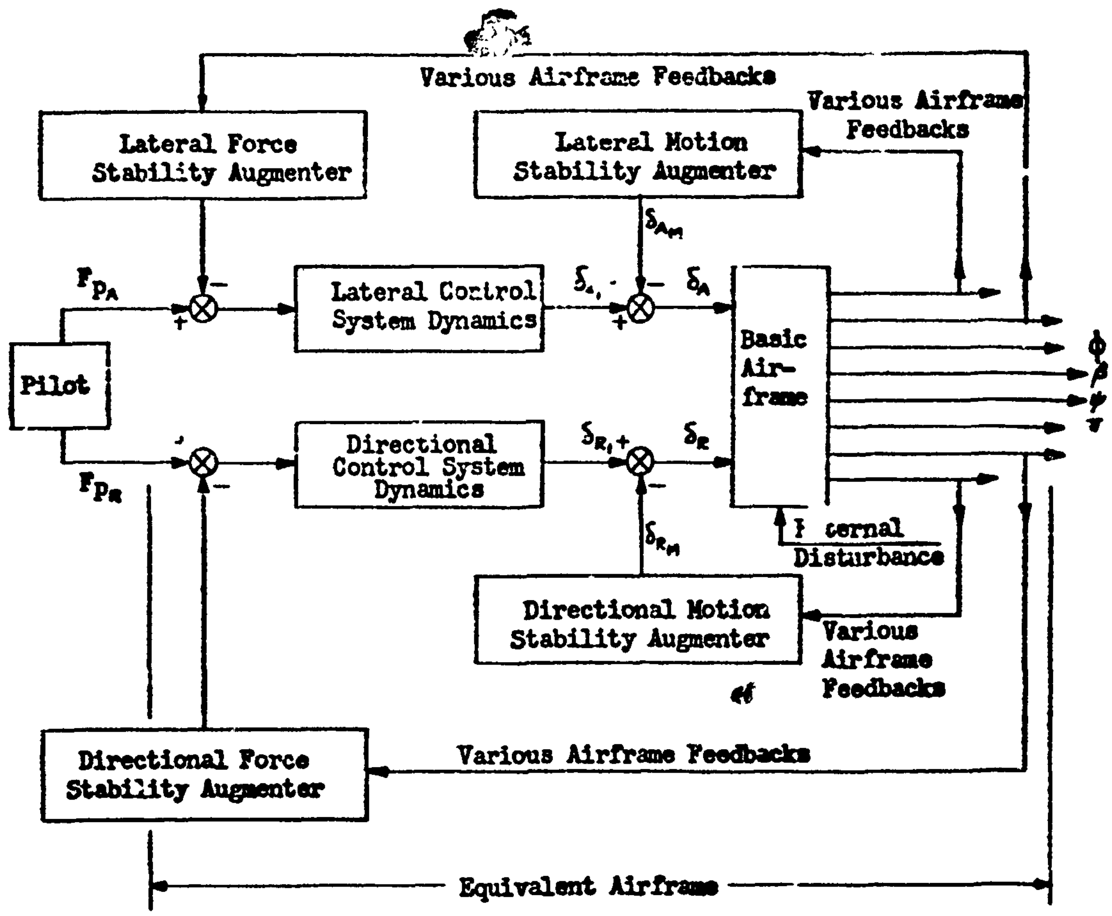

Figure IV-7. Block Diagram of Pilot-Equivalent Airframe (Lateral-Directional) System

(a) DYNAMIC STABILITY

Both the lateral and the directional axes are included in the block diagram because of the close coupling effects of the two axes; i.e., a roll can set up a yaw and vice versa. The Dutch roll oscillation is an example of a condition in which the airplane exhibits oscillations in roll, yaw, and sideslip, all at the same time. Any sustained Dutch roll oscillations are undesirable because the lateral accelerations may become uncomfortable. Furthermore, controlled maneuvers are made more difficult if the airplane goes into Dutch roll oscillations each time a control force, F_{P_A} or F_{P_R}, is applied.

Extensive investigations in the last few years have indicated that pilots prefer a higher degree of damping than is currently specified.* Note that the specifications of References 7 and 9 depend on certain Dutch roll parameters. Reference 9 uses roll to sideslip angle and roll to yaw angle ratios as parameters. Using these parameters, a definite boundary between unsatisfactory and satisfactory Dutch roll damping was established for one given flight condition. To account for flight at different speeds and altitudes, Reference 7 uses roll angle to equivalent side velocity as a parameter.

One point to be noted is that regardless of the damping there is an upper limit on roll to side velocity ratio. It was found that a ϕ/v_e ratio of .55 deg/fps or less was completely satisfactory to the pilots; a ϕ/v_e ratio of .75 deg/fps or less was only tolerable; and a ϕ/v_e ratio greater than .75 deg/fps was intolerable.

An important consideration must be examined at this point. It has been found practical and desirable to eliminate sideslip due to external disturbances by using automatic stability augmentation. When sideslip is eliminated, the so-called Dutch roll oscillation no longer exists, invalidating the graph in Figure IV-1 and also the use of ϕ/v_e and ϕ/β ratios as parameters in specifying lateral-directional dynamic stability.

In the event that Dutch roll is eliminated by reducing sideslip due to external disturbances, a new set of dynamic stability requirements should be specified. These requirements are that the damping of the roll and yaw oscillations should be greater than, or equal to, 0.60. Preferably deadbeat rolls into turns should follow a command input.

The spiral stability requirement is intended to aid the pilot in flying a steady course. During extensive instrument flying or when the pilot must read

* See Figure IV-1, which is presented in Table IV-2 at end of this chapter.

maps, work navigation problems, consult radio facilities handbooks, etc., it is impossible to keep the airplane from diverging spirally. To aid the pilot in his task of keeping the spiral divergence to a minimum, 20 seconds has been suggested as an acceptable time limit for the spiral motion to double amplitude rather than the 4 seconds presently specified. An even more desirable characteristic would be to have an equivalent airframe that does not exhibit a spiral divergence.

(b) STATIC DIRECTIONAL STABILITY

The first static directional stability requirement states that a right rudder pedal force, resulting in a right rudder deflection, should give a left sideslip; i.e., pushing on the right pedal should tend to produce a directional change to the right (see Figure IV-8). This is a desirable characteristic since the direction of control motion corresponds to the resulting direction of response.

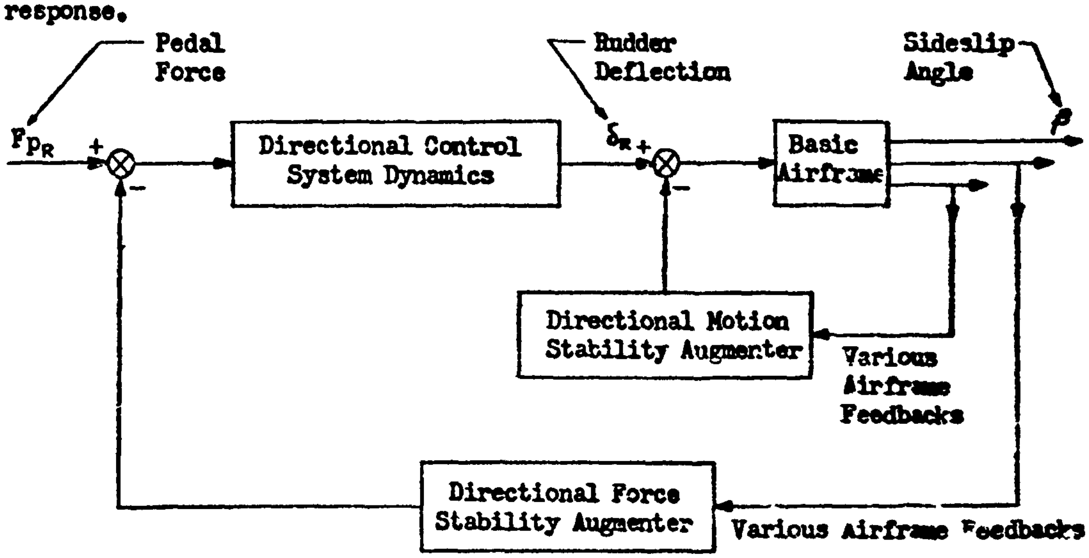

Figure IV-8. Block Diagram Used to Illustrate Static Directional Stability

Furthermore, when the rudder pedal force is released, the airplane should tend to return to a zero sideslip attitude.

The adverse yaw requirement, if met, will tend to simplify the maneuver of making coordinated turns. When rolling into a turn, a yawing moment, due to the aileron deflection and to the inclination of the lift vectors on the wings, is developed which tends to make the downgoing wing move forward. Consequently, in a right roll and turn, the yawing moment tends to move the nose left, or it produces a right sideslip. The pilot must then apply right rudder to offset this adverse yaw effect and reduce the sideslip to zero. Obviously, high rudder fixed directional stability, i.e., small sideslip due to aileron, will make coordinated turns easier. Here again, the use of a sideslip stability augmenter will reduce the adverse yaw effect.

(c) DIHEDRAL EFFECT

Positive dihedral effect is a phenomenon in which left rolls are produced by steady right sideslips; i.e., the leading wing is tipped up. In order to keep the wings level in sideslips, it is required that aileron control deflection and force be directed toward the leading wing. Stated in another way, aileron control deflection and force toward the right should produce right rolls (see Figure IV-9).

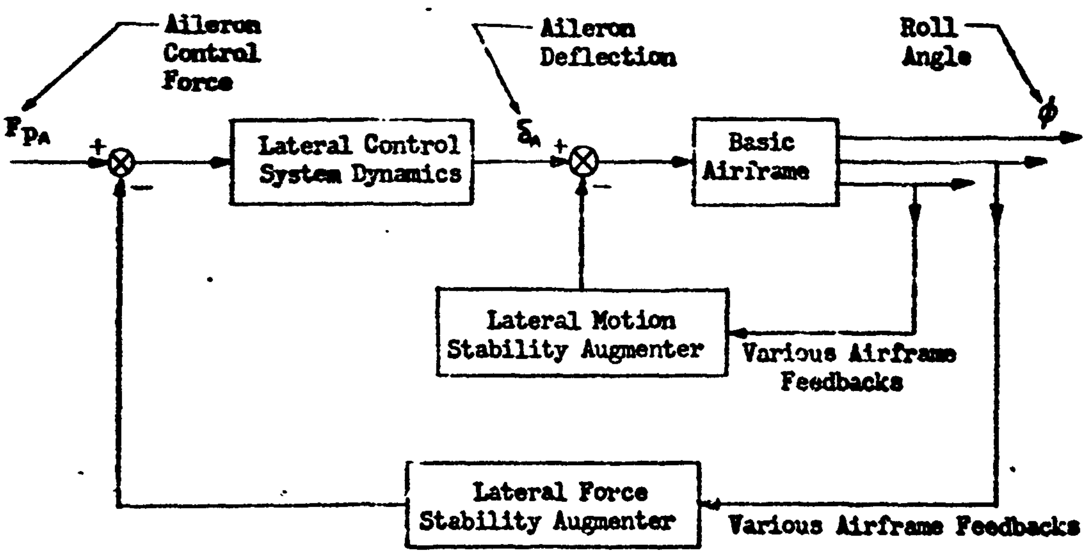

Figure IV-9. Block Diagram Used to Illustrate Dihedral Effect

As with the adverse yaw requirement, the rolling velocity requirement will also simplify coordinated turn maneuvers. Here again, the elimination of sideslip will make this requirement superfluous. However, for completeness, the rolling velocity specification should be examined.

After the ailerons are deflected and a roll is initiated, say, toward the right, the adverse yaw effect will tend to turn the nose to the left; i.e., a right roll will produce a right sideslip. If the adverse yaw effect is great enough, the right sideslip will be quite pronounced. The positive dihedral effect will now tend to roll the airplane away from the sideslip, i.e., tip the right wing up and create a left roll, or a rolling velocity reversal.

Obviously, coordinated maneuvers are very greatly simplified when the adverse yaw and rolling velocity reversal (positive dihedral) effects are minimized. On the other hand, negative dihedral effect is not desirable because negative dihedral effect will continually tend to increase the sideslip; e.g., right sideslip will produce a right roll due to negative dihedral effect; the adverse yaw effect will tend to produce more right sideslip and so on. In other words, negative dihedral effect tends to aggravate the spiral divergence.

(d) RUDDER AND AILERON CONTROL EFFECTIVENESS

The requirements for control effectiveness must be met if the pilot-airplane system is to be able to maneuver properly to accomplish any specific mission. The values specified in Table IV-2 are the minimum required for adequate maneuvering control.

Besides supplying maneuvering control, the rudder and aileron controls must be capable of maintaining steady flight attitudes in any flight configuration. These requirements are specified in Table IV-2.

(e) RUDDER AND AILERON CONTROL FORCES

The control force requirements for the lateral-directional controls are specified mainly to meet the capabilities of the pilot, e.g., the 180 pound

upper limit on the rudder pedal force is approximately 90% of the maximum force that the average pilot can exert. Unlike the longitudinal case, there is not much emphasis put on the force requirements from the standpoint of airframe safety.

(f) RUDDER AND AILERON TRIMMING DEVICES

As with the longitudinal trimming devices, the lateral-directional trimming devices must be capable of reducing the control forces to zero and of maintaining a given setting indefinitely.

(g) APPARENT RUDDER AND AILERON CONTROL SYSTEM FRICTION

Rather than specifying the friction force in the control circuits, the maximum allowable out-of-trim settings due to these forces are specified. This allows for good centering characteristics for both the rudder and aileron.

SECTION 5 - SUGGESTIONS FOR FURTHER STUDY

Most of the specifications presented in the preceding sections were based on subsonic flight investigations. The applicability of these specifications for transonic and supersonic flight must be examined.

Some points of interest to be examined, regardless of the speed regime, are the transient feel problem, cockpit control displacements, and the use of a motion stability augmenter to reduce sideslip from external causes to zero, thus reducing the adverse yaw and rolling velocity reversal effects.

Returning again to the question of flight through different speed regimes, i.e., subsonic, transonic, and supersonic, the aerodynamic behavior of the airplane varies widely through each of these regions. This variation is evidenced even in flight at different Mach numbers and altitudes in one speed regime, as can be seen in Figures III-18, III-28, III-36, and III-57.

This variation in aerodynamic behavior leads naturally to changes in the dynamic and static stability of the airplane and to changes in the control

forces and displacements necessary to perform certain maneuvers. The over-all effect is that the pilot must learn each pattern of control feel cues as he passes from one speed range to another. To say the least, this is confusing to the pilot and complicates his operational procedures.

It is not difficult to visualize the increase in pilot efficiency and improvement in system performance if the equivalent airframe were made invariant for a large range of flight configurations; that is, for any control force or displacement applied by the pilot on the cockpit control, the system response should be the same regardless of the flight condition.

At first glance, this requirement seems rather formidable. However, the results obtained in Chapter III indicate that this invariant applied control-system response relationship can be met by prudent system design. The desirability of establishing this uniform control feel response criterion for pilot-airframe systems should be a field for future study.

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9. Bull, Gifford, "A Flight Investigation of Acceptable Roll to Yaw Ratio of the Dutch Roll, and Acceptable Spiral Divergence," Cornell Aeronautical Laboratory, Report No. TM-574-F-6, Cornell Research Foundation, Inc., Buffalo, New York, February 12, 1952. (Confidential)

10. Williams, W. C., and Crossfield, A. S., "Handling Qualities of High-Speed Airplanes," NACA Research Memorandum, RM L52A08, Langley Aeronautical Laboratory, Langley Field, Virginia, January 28, 1952. (Confidential)

CHARACTERISTIC	FACTORS TO BE CONSIDERED	REQUIREMENT	REF.
DYNAMIC STABILITY	1,2,3, 4,5	Short period oscillations of normal acceleration must be heavily damped for all flight configurations. Short period damping ratio, ζ_{sp}, should be $\zeta_{sp} \geq 0.60$ for optimum design.	1,2
	1, 3	Phugoid oscillations should not be divergent. Preferably, the phugoid damping ratio, ζ_p, should be $\zeta_p \geq$ 0.20 for optimum design.	3
STATIC STABILITY	4, 5	Stick-fixed and stick-free static stability for all flight configurations is required such that increasing up elevator will reduce trim speed and such that with elevator free, the airplane will tend to return to trim speed following a disturbance.	1,2
ELEVATOR CONTROL EFFECTIVENESS Trim	5	Elevator control must be sufficient to obtain and maintain steady flight at all flight configurations.	1,2
Manoeuvers	4	Elevator control must be powerful enough to develop maximum positive or negative lift coefficient, or positive or negative limit load factor for all flight configurations.	1,2
Landing	4, 5	Elevator control must be sufficient to hold the airplane off but very near the ground at near stall conditions with the most forward center of gravity position.	1,2
Take-Off	4	Elevator control must be sufficient to maintain the airplane at any ground attitude up to take-off attitude with greatest tail-heavy weight moment about the main wheels for tail-wheel airplanes or greatest nose-down static weight moment about the main wheels for nose-wheel airplanes.	1,2
ELEVATOR CONTROL FORCES	2,3,4	Elevator control forces for the above conditions should not be exceedingly high nor should they be too low. Increasing pull-forces should produce increases in normal acceleration.	1,2
Manoeuvers: Steady Turns and Steady Pull-Ups	2,3,4	The elevator control force vs. change in normal acceleration factor curve should have a slope within the following ranges for all flight configurations and weight loadings: $3 \leq 21/(n-1) < F/g < 56/(n-1)$ lbs/g for class I, III, and IV airplanes* $3 \leq 45/(n-1) < F/g < 120/(n-1)$ lbs/g for class II airplanes* (n is design limit load factor in g units)	1,2
Manoeuvers: Sudden Pull-Ups	3	The gradient of maximum elevator control force with maximum normal acceleration in sudden pull-ups from trimmed straight flight should be greater than or equal to F/g specified above.	1,2
Landing	2,3,4	The elevator control pull force required to meet the landing requirement should be greater than 15 lbs at the most aft center of gravity position and less than 35 lbs at the most forward center of gravity position for class I and III and for all carrier based airplanes. For all other airplanes, the maximum pull force required should be less than 50 lbs.	1,2,3
Take-Off	2,3,4	The elevator control forces for the take-off requirements should be less than: 20 lbs push for class I and III and for all carrier based airplanes } (tail-wheel) 35 lbs push for all other airplanes and 35 lbs pull for lift-off for class I and III and for all carrier based airplanes } (nose-wheel) 50 lbs pull for lift-off for all other airplanes	1,2
TRIM	2,3,5	Trim changes at any speed due to any combination of power, flap, or landing gear settings should be minimized or zero if possible. There should be no reversal of trim. The elevator control force required to counteract any trim change should be less than 20 lbs for class I, III, or IV airplanes or 30 lbs for class II airplanes.	1,2,3
	2,5	There should be substantially no trim change for less than 10° of sideslip, and no more than 10 lbs of pull-force should be required for trim at any sideslip angle which can be produced with 50 lbs of pedal force.	1,2,3
	5	Any steady state error following a disturbance should not exceed ± 2 mph in trim speed or ±.25° in flight path angle when the aircraft is trimmed for straight flight and the center of gravity is half-way between the most forward and most aft positions.	3
TRIMMING DEVICES	5	The trimming device should maintain a given setting indefinitely unless changed intentionally.	1,2
	5	The trimming device should be capable of reducing the elevator control force to zero.	1,2

Table IV-1. Codification of Longitudinal Equivalent Airframe Stability and Control Requirements

* Class I - Light Airplanes
Class II - Patrol, heavy attack, transport, cargo, etc.
Class III - Fighters, attack, dive bombers, etc. (shore based)
Class IV - Fighters, general-purpose attack, etc. (ship based)

CHARACTERISTIC	FACTORS TO BE CONSIDERED	REQUIREMENT	REF.
DYNAMIC STABILITY Dutch Roll	1,4,5	The lateral-directional oscillation (Dutch roll) should be stable. The damping should be in accordance with Figure IV-2 for all flight configurations.	1,2,3, 7,3,9

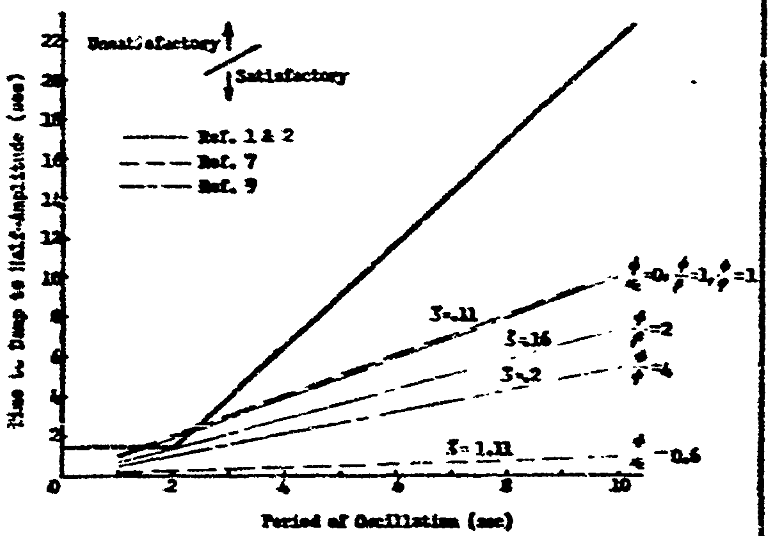

Figure IV-2. Boundary Between Satisfactory and Unsatisfactory Lateral-Directional Oscillatory Characteristics

CHARACTERISTIC	FACTORS TO BE CONSIDERED	REQUIREMENT	REF.
Spiral Mode	2, 5	The spiral divergence should not double amplitude in less than 20 seconds for all flight conditions.	9
STATIC DIRECTIONAL STABILITY	4, 5	Rudder-fixed and rudder-free static directional stability should be such that increasing right rudder deflection increases left sideslip and such that with rudders free the airplane will always tend to return to trim conditions with wings level from steady sideslips.	1, 2
Adverse Yaw Requirement	4	Rudder-fixed static directional stability should be such that the angle of sideslip due to sudden aileron deflection is limited to less than 5° of sideslip for full aileron deflection when rolling out of trimmed steady 45° banked turns.	1,2,3
	5	Rudder-free static directional stability on multi-engine planes should be such that with any one engine inoperative, the airplane with rudder free may be balanced directionally in steady straight flight by sideslipping and banking.	1, 2
DIHEDRAL EFFECT	4, 5	Aileron-fixed and aileron-free positive dihedral effect is required such that aileron control deflection and aileron control force toward the leading wing is necessary to trim in steady sideslips.	1, 2
Rolling Velocity Reversal Requirement	4, 5	The dihedral effect should not be of such a nature that an excessively large rolling moment due to sideslip is developed, thus causing a reversal of rolling velocity due to adverse yaw during rudder-free aileron rolls.	1, 2
	4, 5	Negative dihedral effect which will increase any steady sideslip is not desirable.	1, 2
	4, 5	The variation of side force with angle of sideslip should be such that right skidding turns accompany right rudder deflection (and vice versa) when the wings are held level and the rudder is deflected from the position required for straight flight.	1, 2

CHARACTERISTIC

- RUDDER CONTROL EFFECTIVENESS
- AILERON CONTROL EFFECTIVENESS
- RUDDER AND AILERON CONTROL FORCES
- RUDDER AND AILERON TRIMMING DEVICES
- APPARENT CONTROL SYSTEM FRICTION

Table IV-2. Codification of Lateral-Directional Equivalent Airframe Stabil

	REF.
be stable. The damping configurations.	1,2,3, 7,8,9
	
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and Unsatisfactory y Characteristics	
less than 20 seconds for all	9
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ch that the angle of side- less than 5° of sideslip for teady 45° banked turns.	1,2,3
ine planes should be such th rudder free may be bal- slipping and banking.	1, 2
ct is required such that toward the leading wing is	1, 2
at an excessively large sing a reversal of rolling n rolls.	1, 2
y sideslip is not desirable.	1, 2
ould be such that right side- e versa) when the wings are ion required for straight	1, 2

CHARACTERISTIC	[illegible] TO BE [illegible]	REQUIREMENT	REF.
RUDDER CONTROL EFFECTIVENESS	3 5	The rudder should give sufficient directional control to balance the airplane in steady straight flight with the wings level in all configurations.	1, 2
	4, 5 5	In the landing configuration, full rudder deflection should produce at least 10° of steady sideslip.	1, 2
	5 5	The rudder control in conjunction with other means of control should be adequate to maintain straight paths on the ground during take-offs and landings in cross-winds and in normal conditions.	1, 2
	5 5	The rudder control in conjunction with other means of control should be adequate for taxiing on land and water.	1, 2
	5 5	The rudder control on multi-engine airplanes should be adequate to hold the airplane with a zero yawing velocity and not more than 5° bank with any one engine inoperative.	1, 2
	4, 5 5	The rudder should be capable of overcoming the adverse yaw when the airplane is rolled abruptly out of steady 45° banked turns using full aileron deflection.	1, 2
AILERON CONTROL EFFECTIVENESS	4, 5		
	4, 5	Positive aileron deflection (right aileron up, left aileron down) should always give right bank.	1, 2
	4 4	The peak value for rolling acceleration should occur in less than 3.75 c/V seconds or 0.1 second, whichever is the greater, after full aileron deflection is reached in a rudder-locked roll.	1, 2
	4 4	The maximum rolling velocity obtained by a rudder-fixed aileron deflection should be approximately proportional to the aileron deflection from trim.	1, 2
	4 4	In rolls from straight or turning flights with rudder fixed, the aileron should be sufficiently powerful to produce a bank angle of 90° within 1 second after the aileron is deflected.	10
	5 5	The ailerons should be powerful enough to keep the wings level when any one engine is inoperative in a multi-engine airplane.	1, 2
RUDDER AND AILERON CONTROL FORCES	2 2	The rudder pedal forces for any flight configuration and maneuver should not exceed 180 lbs.	1, 2
	2 2	The maximum aileron control force for the bank angle requirement should not exceed the following: Class I, III, IV 30 lbs stick force or 60 lbs wheel force Class II 30 lbs stick force or 80 lbs wheel force	1, 2
	2 2	The aileron control force to meet the asymmetric power requirement should not exceed the forces specified above.	1, 2
	5 5	The aileron control force vs. stick or wheel deflection curve should be smooth and steep enough to return the control to approximate trim position when released.	1, 2
RUDDER AND AILERON TRIMMING DEVICES	5 5	The trimming device should maintain a given setting indefinitely unless changed intentionally.	1, 2
	5 5	The trimming device should be capable of reducing the elevator control force to zero.	1, 2
APPARENT CONTROL SYSTEM FRICTION	5 5	The amount of apparent rudder and aileron control system friction should not be so great as to cause the controls to stick to more than 1° of total aileron or 2° of rudder from trim position after having been applied and slowly released at approach speed.	3

-2. Codification of Lateral-Directional Equivalent Airframe Stability and Control Requirements

2.

APPENDIX

DERIVATION OF AUGMENTED CHARACTERISTIC EQUATION OF THE EQUIVALENT AIRFRAME

The equations of motion of the basic airframe are

$$(A\text{-}1)\qquad \begin{cases} \dot{u} = X_u u + X_w w - g\theta \\ \dot{w} = Z_u u + Z_w w + (U_0 + Z_{\dot{\theta}})\dot{\theta} + Z_{\delta_e}\delta_e \\ \ddot{\theta} = M_u u + M_w w + M_{\dot{w}}\dot{w} + M_{\dot{\theta}}\dot{\theta} + M_{\delta_e}\delta_e \\ a_z = \dot{w} - U_0\dot{\theta} \end{cases}$$

From (A-1), the transfer functions relating u and a_z to δ_e may be derived. These are*

$$(A\text{-}2)\qquad \begin{cases} \dfrac{u}{\delta_e} = \dfrac{B_u s^2 + C_u s + D_u}{As^4 + Bs^3 + Cs^2 + Ds + E} \\[2ex] \dfrac{a_z}{\delta_e} = \dfrac{s(A_{a_z} s^3 + B_{a_z} s^2 + C_{a_z} s + D_{a_z})}{As^4 + Bs^3 + Cs^2 + Ds + E} \end{cases}$$

where

$$B_u \cong Z_{\delta_e} X_w$$

$$C_u \cong -Z_{\delta_e}(gM_{\dot{w}} + M_{\dot{\theta}} X_w) + M_{\delta_e}(U_0 X_w - g)$$

$$D_u \cong g(M_{\delta_e} Z_w - Z_{\delta_e} M_w)$$

$$A_{a_z} \cong Z_{\delta_e}$$

$$B_{a_z} \cong M_{\delta_e} Z_{\dot{\theta}} - Z_{\delta_e} M_{\dot{\theta}}$$

$$C_{a_z} \cong M_{\delta_e} U_0 Z_w - Z_{\delta_e}(U_0 M_w - X_u M_{\dot{\theta}})$$

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* See Equations (III-1), (III-4), and (III-29) of Methods of Analysis and Synthesis of Piloted Aircraft Flight Control Systems, BuAer Report No. AE-61-4 I, Northrop Aircraft, Inc., Hawthorne, California, March 1952.

$$D_{\delta_E} \cong -M_{\delta_E}\left[Z_u g + U_o(Z_w X_u - X_w Z_u)\right] + Z_{\delta_E}\left[M_u g + U_o(M_w X_u - M_u X_w)\right]$$

$$A = 1$$

$$B \cong -U_o M_{\dot{w}} - M_{\dot{\theta}} - Z_w$$

$$C \cong M_{\dot{\theta}} Z_w - U_o M_w$$

$$D \cong -X_u(M_{\dot{\theta}} Z_w - U_o M_w) - M_u(X_w U_o - g)$$

$$E \cong g(Z_u M_w - M_u Z_w)$$

Consider now u/δ_E in (A-2), letting

(A-3) $$\delta_E = \delta_{E_p} + (K_u + K_{\dot{u}} s)u$$

where

δ_{E_p} is the elevator motion caused by the pilot

K_u is the amount of u feedback to the elevator through the force and motion stability augmenters

$K_{\dot{u}}$ is the amount of $\dot{u}$ feedback to the elevator through the force and motion stability augmenters

then

(A-4) $$u = \frac{B_u s^2 + C_u s + D_u}{As^4 + Bs^3 + Cs^2 + Ds + E}\left[\delta_{E_p} + (K_u + K_{\dot{u}} s)u\right]$$

or

(A-5) $$\frac{u}{\delta_{E_p}} = \frac{B_u s^2 + C_u s + D_u}{(As^4 + Bs^3 + Cs^2 + Ds + E) - (K_u + K_{\dot{u}} s)(B_u s^2 + C_u s + D_u)}$$

(A-5)
(cont)

$$= \frac{B_u s^2 + C_u s + D_u}{As^4 + (B - K_{\dot u} B_u)s^3 + (C - K_u B_u - K_{\dot u} C_u)s^2 + (D - K_u C_u - K_{\dot u} D_u)s + (E - K_u D_u)}$$

$$= \frac{B_u s^2 + C_u s + D_u}{As^4 + (B + \Delta B_{\dot u})s^3 + (C + \Delta C_u + \Delta C_{\dot u})s^2 + (D + \Delta D_u + \Delta D_{\dot u})s + (E + \Delta E_u)}$$

From (A-5) and (A-2),

(A-6)*
$$\left[\begin{aligned} \Delta C_u &= -K_u B_u \approx -K_u Z_{\delta_e} X_w \\ \Delta D_u &= -K_u C_u \approx K_u\left[Z_{\delta_e}(gM_{\dot w} + M_q X_w) - M_{\delta_e}(U_0 X_w - g)\right] \\ \Delta E_u &= -K_u D_u \approx gK_u(Z_{\delta_e} M_w - M_{\delta_e} Z_w)^{*} \end{aligned}\right.$$

and

(A-7)**
$$\left[\begin{aligned} \Delta B_{\dot u} &= -K_{\dot u} B_u = -K_{\dot u} Z_{\delta_e} X_w \\ \Delta C_{\dot u} &= -K_{\dot u} C_u = K_{\dot u}\left[Z_{\delta_e}(gM_{\dot w} + M_q X_w) - M_{\delta_e}(U_0 X_w - g)\right] \\ \Delta D_{\dot u} &= -K_{\dot u} D_u = K_{\dot u} g(Z_{\delta_e} M_w - M_{\delta_e} Z_w) \end{aligned}\right.$$

Now consider a_z/δ_e in (A-2), letting

(A-8) $$\delta_p = \delta_{e_p} + (K_{a_z} + K_{\dot a_z} s)a_z$$

where

K_{a_z} is the amount of a_z feedback to the elevator through the force and motion stability augmenters

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* See (III-28).

** See (III-20).

$K_{\dot{a}_x}$ is the amount of $\dot{a}_x$ feedback to the elevator through the for- nd motion stability augmenters.

Then

$$\text{(A-9)}\qquad a_x = \frac{s(A_{a_x}s^3 + B_{a_x}s^2 + C_{a_x}s + D_{a_x})}{As^4 + Bs^3 + Cs^2 + Ds + E}\left[\delta_{e_p} + (K_{a_x} + K_{\dot{a}_x}s)a_x\right]$$

or

$$\text{(A-10)}\qquad \frac{a_x}{\delta_{e_p}} = \frac{s(A_{a_x}s^3 + B_{a_x}s^2 + C_{a_x}s + D_{a_x})}{(As^4 + Bs^3 + Cs^2 + Ds + E) - (K_{a_x} + K_{\dot{a}_x}s)s(A_{a_x}s^3 + B_{a_x}s^2 + C_{a_x}s + D_{a_x})}$$

$$= \frac{s(A_{a_x}s^3 + B_{a_x}s^2 + C_{a_x}s + D_{a_x})}{-K_{\dot{a}_x}A_{a_x}s^5 + (A - K_{a_x}A_{a_x} - K_{\dot{a}_x}B_{a_x})s^4 + (B - K_{a_x}B_{a_x} - K_{\dot{a}_x}C_{a_x})s^3 + (C - K_{a_x}C_{a_x} - K_{\dot{a}_x}D_{a_x})s^2 + (D - K_{a_x}D_{a_x})s + E}$$

$$= \frac{s(A_{a_x}s^3 + B_{a_x}s^2 + C_{a_x}s + D_{a_x})}{A_1s^5 + (A + \Delta A_{a_x} + \Delta A_{\dot{a}_x})s^4 + (B + \Delta B_{a_x} + \Delta B_{\dot{a}_x})s^3 + (C + \Delta C_{a_x} + \Delta C_{\dot{a}_x})s^2 + (D + \Delta D_{a_x})s + E}$$

From (A-10) and (A-2),

$$\text{(A-11)*}\qquad \left[\begin{aligned}
\Delta A_{a_x} &= -K_{a_x}A_{a_x} = K_{a_x}Z_{\delta_e} \\
\Delta B_{a_x} &= -K_{a_x}B_{a_x} = K_{a_x}(Z_{\delta_e}M_{\dot{\theta}} - M_{\delta_e}Z_{\dot{\theta}}) \\
\Delta C_{a_x} &= -K_{a_x}C_{a_x} = K_{a_x}\left\{\left[Z_{\delta_e}(U_0M_w - X_uM_{\dot{\theta}})\right] - M_{\delta_e}U_0Z_w\right\} \\
\Delta D_{a_x} &= -K_{a_x}D_{a_x} = K_{a_x}\left\{Z_{\delta_e}\left[U_0(M_uX_w - M_wX_u) - M_ug\right] + M_{\delta_e}\left[U_0(Z_wX_u - X_wZ_u) + Z_ug\right]\right\}
\end{aligned}\right.$$

and

$$\text{(A-12)**}\qquad \left[\begin{aligned}
A_1 &= -K_{\dot{a}_x}Z_{\delta_e} \\
\Delta A_{\dot{a}_x} &= K_{\dot{a}_x}(Z_{\delta_e}M_{\dot{\theta}} - M_{\delta_e}Z_{\dot{\theta}}) \\
\Delta B_{\dot{a}_x} &= K_{\dot{a}_x}\left\{\left[Z_{\delta_e}(U_0M_w - X_uM_{\dot{\theta}})\right] - M_{\delta_e}U_0Z_w\right\} \\
\Delta C_{\dot{a}_x} &= K_{\dot{a}_x}\left\{Z_{\delta_e}\left[U_0(M_uX_w - M_wX_u) - M_ug\right] + M_{\delta_e}\left[U_0(Z_wX_u - Z_uX_w) + Z_ug\right]\right\}
\end{aligned}\right.$$

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* See (III-33)
** See (III-34)